(NASA-CR-161564) ORBIT TRANSFER VEHICLE ENGINE STUDY. VOLUME 2: TECHNICAL REPORT Final Report (Pratt and Whitney Aircraft) 206 p HC A10/MF A01 CSCL 21H

N80-32432

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FR-13158
VOLUME II
31 July 1980

FINAL REPORT

ORBIT TRANSFER VEHICLE ENGINE STUDY

TECHNICAL REPORT

Contract NAS8-33444

Prepared for
National Aeronautics and Space Administration
George C. Marshall Space Flight Center
Marshall Space Flight Center, Alabama 35812

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FOREWORD

This technical report presents the results of the Orbit Transfer Vehicle Engine Study. The study was conducted by the Pratt & Whitney Aircraft Group, Government Products Division of the United Technologies Corporation for the National Aeronautics and Space Administration's George C. Marshall Space Flight Center under contract NAS8-33444.

The results of the study are contained in the following three volumes which are submitted in accordance with the data requirements of Contract NAS8-33444.

Volume I -- Executive Summary
Volume II -- Final Technical Report
Volume III -- Program Costs

This study was initiated in July 1979 with the technical effort completed in eight months and the final report delivered April 1980. The study effort was conducted under the direction of the George C. Marshall Space Flight Center Science and Engineering Organization with Mr. Dale H. Blount as Contracting Officer's Representative. The effort at P&WA/GPD was carried out under the direction of James R. Brown, Program Manager.

The following individuals have provided significant contributions in the preparation of this report.

Charles D. Limerick -- Systems Performance Analysis
Donald E. Galler -- Engine Cycle Analysis
Philip S. Thompson -- Programatics and Cost Analysis

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SECTION 1

INTRODUCTION

The objective of the Orbit Transfer Vehicle (OTV) engine study was to provide parametric performance, engine programmatic, and cost data on the complete propulsive spectrum that is available for a variety of high energy, space-maneuvering missions. The OTV is planned as a high-performance propulsive stage which can be used, in conjunction with the Space Shuttle, to deliver/support large payloads platforms to geosynchronous earth orbit (GEO) and other orbits beyond low earth orbit (LEO). Its role is similar to that of the "full capability" Space Tug defined in 1974 with the primary difference that the OTV will eventually be man-rated. This engine study was conducted to satisfy a set of technical requirements derived around the OTV system to provide both engine system parametric data and preliminary engine designs.

This study addresses the propulsion system spectrum by covering candidate OTV engines from the near-term RL10 (and its derivatives) to advanced high-performance expander and staged-combustion cycle engines. A study plan flow diagram identifying program objectives is shown in Figure 1-1. The RL10/RL10 derivative performance, cost and schedule data were updated and provisions defined which would be necessary to accommodate extended low-thrust operation. Parametric performance, weight, envelope, and cost data were generated for advanced expander and staged-combustion OTV engine concepts. A prepoint design study was conducted to optimize thrust chamber geometry and cooling, engine cycle variations, and controls for an advanced expander engine. Operation at low thrust was defined for the advanced expander engine and the feasibility and design impact of kitting was investigated. An analysis of crew safety and mission reliability was conducted for both the staged-combustion and advanced expander OTV engine candidates.

The schedule followed by Pratt & Whitney Aircraft during the performance of these activities is shown in Figure 1-2. Results of the study are detailed in the following sections of this report. Engine definition and requirements are provided in Section 2, parametric studies are discussed in Section 3, and Sections 4 and 5 present the advanced expander engine optimization and low thrust operation, respectively. Crew safety and engine reliability are discussed in Section 6 with engine program plans and estimated costs in Section 7. Conclusions and recommendations are presented in Sections 8 and 9.

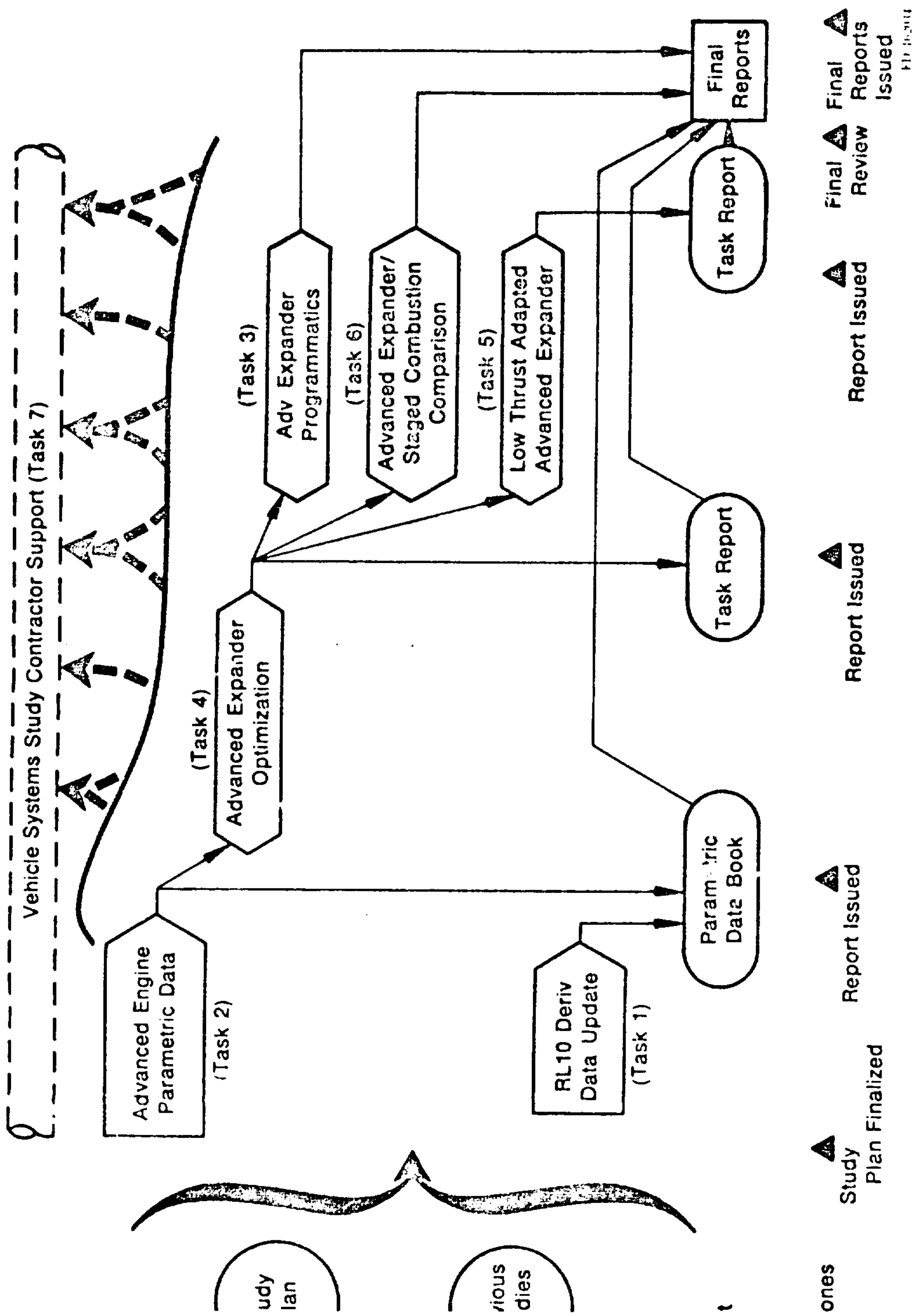


Figure 1-1. Study Plan Summary

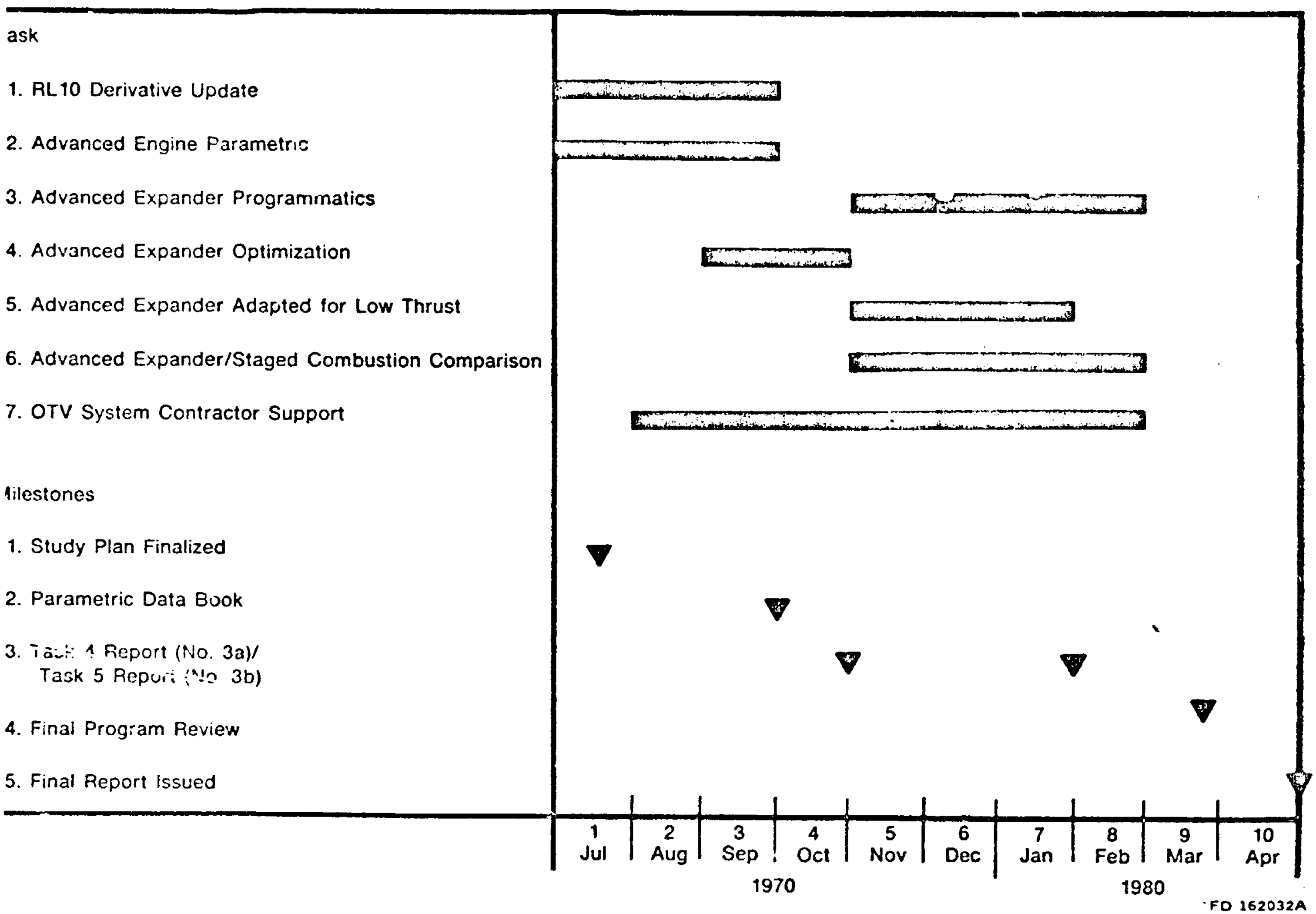


Figure 1-2. Orbit Transfer Vehicle Engine Study Schedule

SECTION 2

BASELINE ENGINES

2.0 GENERAL

Three of the engines defined under Contract NAS8-28989 (the RL10 Derivative IIA, IIB and Category IV) were updated during this study. In addition, the Derivative IIC was added to provide data on a moderately high-performance engine for use in an early operational orbital transfer vehicle (OTV). Further, the Advanced Expander Cycle engine which is a new "1980 state-of-the-art" high performance OTV engine candidate, was added. The primary technical aspect of the RL10 derivative engine updating is related to the predicted specific impulse values for the engines. Since the previous study was completed in 1973, testing with LO_2/LH_2 of high-area-ratio nozzles (the ASE at 175 to 1 and 400 to 1 and the RL10 at 205 to 1) has indicated that the accepted JANNAF procedures (see Figure 2-1) for predicting engine specific impulse are conservative. Figure 2-2 presents a comparison of JANNAF-predicted performance and test data at a mixture ratio of 6 to 1. Figure 2-3 presents this data as a ratio of measured-to-predicted specific impulse and shows an apparent correlation of the difference with nozzle area ratio (approximately a 1.3% error at the 400 to 1 area ratio data point). It is probable that this difference is due to difficulties in accurately predicting boundary-layer loss characteristics of high-area-ratio nozzles. Since the engines of this study all have high area ratio nozzles (up to approximately 900 to 1) it was desirable to use a performance prediction technique which provided a better correlation with the measured data if possible. It was found that replacing the BLIMP nozzle boundary-layer-loss procedure with a simplified procedure for calculating nozzle wall friction loss as a function of Mach number and gas properties produced such a result. This simplified procedure produced results averaging approximately 0.2% greater than the measured data (worst case 0.42%) and did not correlate with nozzle area ratio (see Figure 2-3). This technique was used for all specific impulse predictions of this study including an adjustment of −0.2%.

Additionally, the weight of the updated engines was adjusted to reflect the use of a carbo-composite radiation-cooled extendible nozzle rather than the sheet-metal dump-cooled design used in the previous study. The baseline engine design point characteristics of each engine at 15,000 lb thrust are summarized in Table 2.1.

A general description of the performance and operating characteristics of the baseline RL10 Derivative IIA, IIB, IIC, Category IV and Advanced Expander Cycle engines are given in the following paragraphs. A more detailed description of these engines (except the Derivative IIC and the Advanced Expander Cycle) may be found in P&WA Report FR-6011, "Design Study of RL10 Derivatives, Final Report, 15 December 1972."

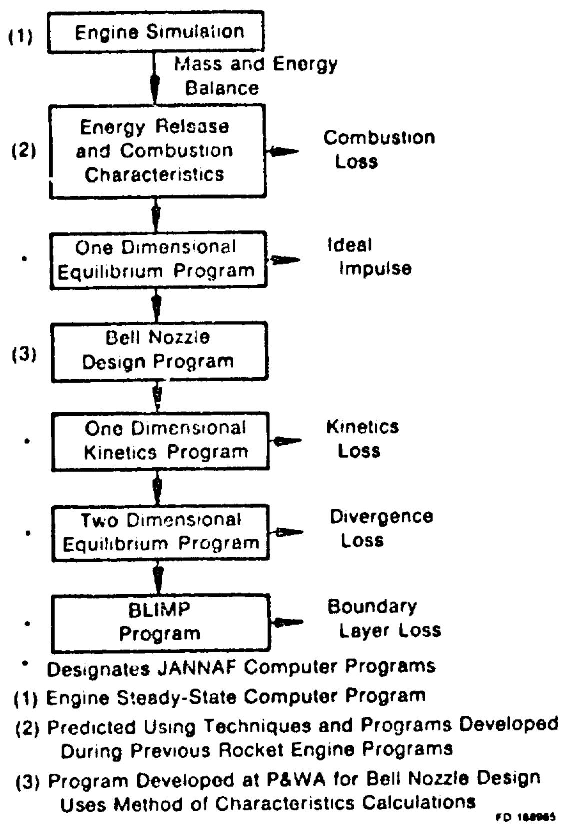
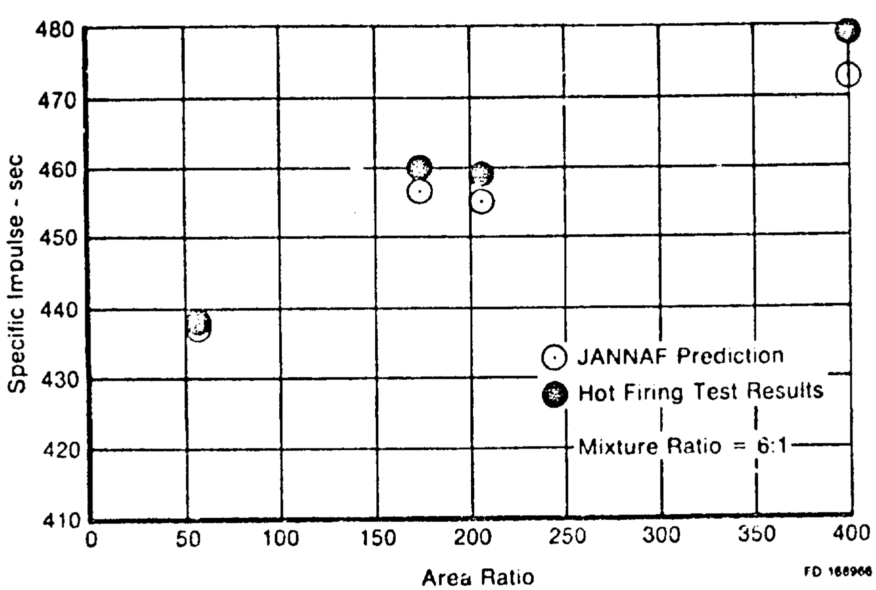


Figure 2-1. Performance Prediction Programs



Figur 2-2. Comparison of JANNAF-Predicted Performance With Test Results

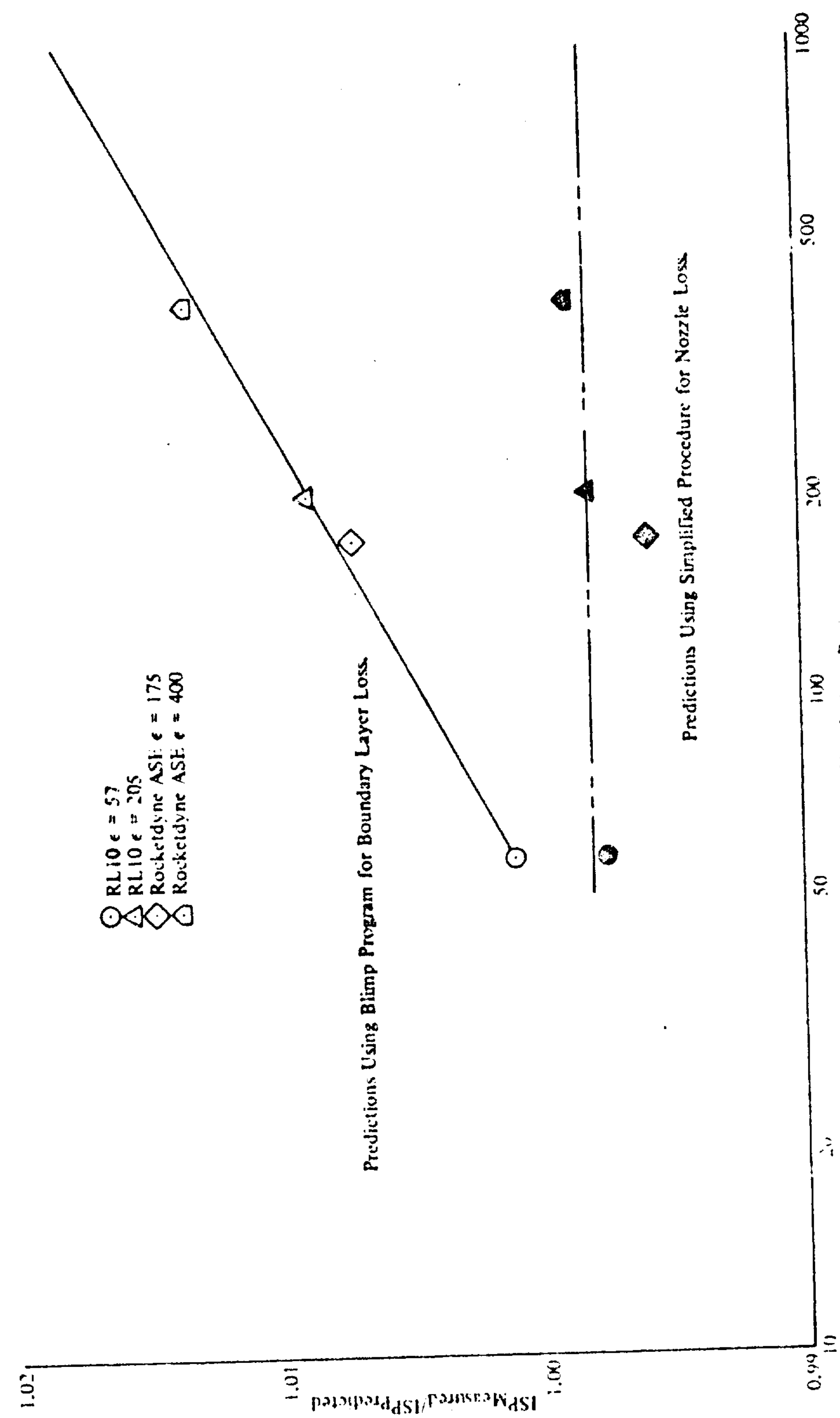


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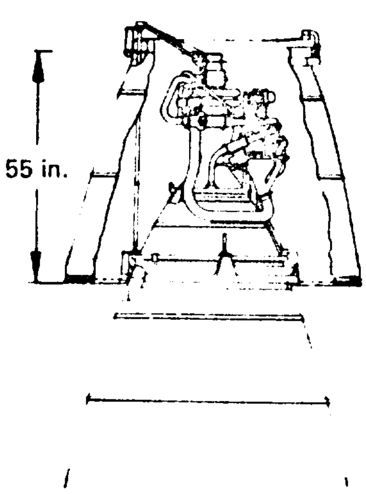
Figure 2-3. Specific Impulse Adjustment for Area Ratio

TABLE 2-1. BASELINE ENGINE DESIGN POINT CHARACTERISTICS

	Derivative IIA	*Derivative IIB*	*Derivative IIC*	*Category IV*	*Advanced Expander*
Full Thrust (vac), lb	15,000	15,000	15,000	15,000	15,000
Mixture Ratio, Nominal	6.0	6.0	6.0	6.0	6.0
Chamber Pressure, psia	400	400	400	915	1505
Specific Impulse, sec	459.8	459.8	458.6	471.7	482.0
Required Inlet Conditions (Full Thrust)					
Fuel, NPSP, psi	0	0.5	2	0	0.5
Oxidizer, NPSP, psi	0	4	4	0	1
Installed Length, in.	55	55	55	55	60
Weight, lb	431	392	374	371	391
Nozzle Area Ratio	205	205	205	388	640
Engine Life, Firings/hr	190/5[1]	190/5[1]	10/1.25[2]	300/10[1]	300/10[1]
Engine Conditioning	Tank-Head Idle	Tank-Head Idle	Overboard Dump Cooldown	Tank-Head Idle	Tank-Head Idle
Maneuvering Thrust Capability (pumped idle)	Yes	Yes	No	Yes	Yes
Development Program					
Time to FFC, Mo.	64	58	37	80	89
Cost, $79M*	100	79	21	157	243

*Including propellant cost, without Fee
1. Time Between Overhauls (TBO)
2. Expendable Mission
3. Design TBO

2.1 RL10 DERIVATIVE IIA ENGINE



Thrust	: 15,000 lb
Chamber Pressure	: 400 psia
Area Ratio	: 205
I_{sp}	: 459.8 sec at 6.0 MR
Operation	: Full Thrust (Saturated Propellants)
	: Maneuver Thrust (Saturated Propellants)
Conditioning	: Tank Head Idle
Weight	: 431 lb
Life (TBO)	: 190 Firings/5 hr
DDT&E Cost	: $100 Million

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2.1.1 Definition and Requirements

The RL10 Derivative IIA engine is derived from the basic RL10A-3-3 but has increased performance and operating flexibility for use in the OTV. With a nominal full thrust level of 15,000 lb (in vacuum) at a mixture ratio of 6.0:1, the Derivative IIA engine is defined as an RL10A-3-3 with the following changes:

1. Two-position nozzle with recontoured primary section to give a large increase in specific impulse with engine installed length no greater than the RL10A-3-3 (70 in.). With a truncated two-position nozzle installed, this engine can be installed and tested in the existing test facilities at P&WA/GPD.

2. Injector reoptimized for operation at a full thrust mixture ratio of 6.0:1.

3. Tank head idle (THI) capabilities, where the engine is run pressure fed without its turbopump rotating on propellants supplied from the vehicle tanks at saturation pressure. Propellant conditions at the engine inlets can vary from superheated vapor, through mixed phase, to liquid. The objectives are to supply low thrust to settle vehicle propellants and also to obtain useful impulse from the propellants used to condition the engine and vehicle feed system.

4. Operation at low thrust in pumped mode (maneuver thrust) but without significant impact on the engine's design. This thrust level was selected as 25% of full thrust in the previous study.

5. Two-phase pumping capability, allowing operation at both full and maneuver thrust levels with saturated propellants in the vehicle tanks and with no tank pressurization system or vehicle-mounted boost pumps.

6. Capability for both H_2 and O_2 autogenous pressurization which may be required on very long burn missions in order to avoid excessively low propellant vapor pressure.

2.1.2 Description

The general arrangement of the RL10 Derivative IIA engine is shown by the installation drawings in Figures 2-4 and 2-5. This engine is interchangeable with the RL10A-3-3 except for the larger diameter of the propellant inlet valves and the shorter distance between the inlet valve interfaces and the engine reference plane.

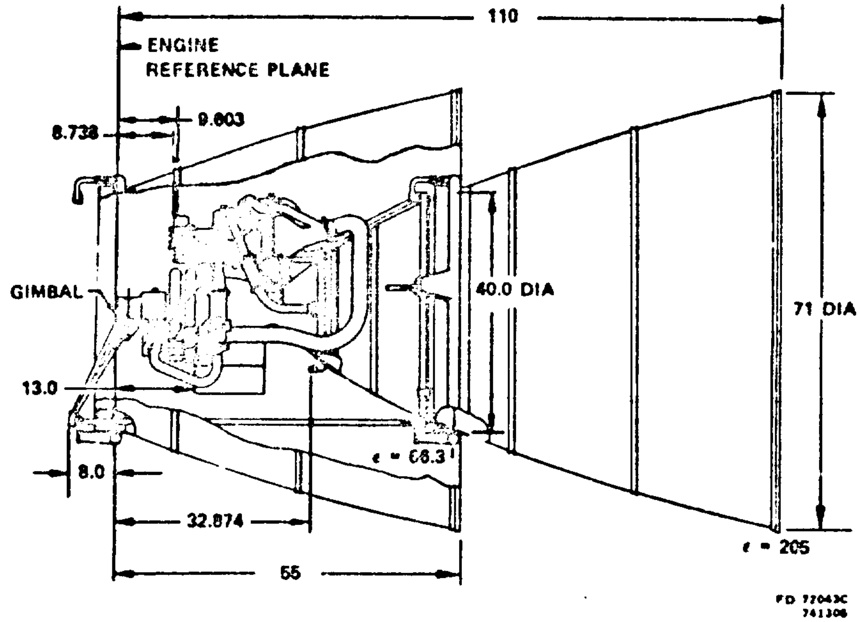


Figure 2-4. Derivative IIA Engine Installation Drawing (Sheet 1)

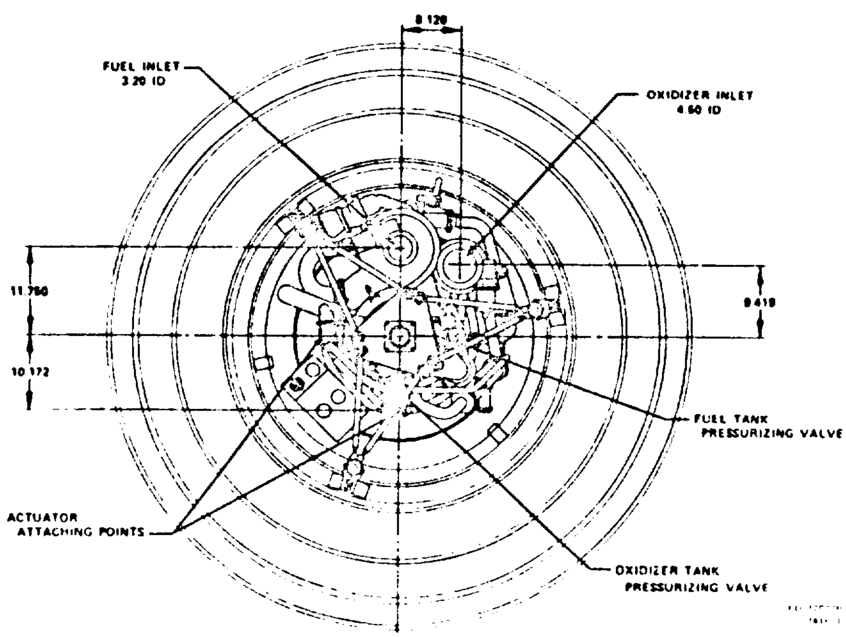


Figure 2-5. Derivative IIA Engine Installation Drawing (Sheet 2)

The principal components of this engine are shown in Figure 2-6. To be able to pump two-phase propellants, larger inlet shutoff valves, a redesigned fuel high-speed inducer (which is based on the RL10 Mod 2 design), and a new gear-driven oxidizer low-speed inducer are required. To avoid changing the location of the engine/vehicle propellant system interfaces, the oxidizer pump and its shaft are rotated through 180 deg. This arrangement is illustrated by Figure 2-7 which compares the RL10A-3-3 and Derivative IIA turbopumps. The fuel pump interstage chilldown valve is deleted, since the engine is conditioned by running in THI mode. A GO_2 heat exchanger, GO_2 control valve and turbine bypass valve are added to enable the engine to run in THI. Fuel and oxidizer tank pressurization valves are added to give autogenous pressurization capability. Additional solenoid valves and modifications to the oxidizer flow control valve and thrust control valve give the engine its capability to operate in three modes. A dual exciter gives improved ignition reliability in THI. The primary nozzle is recontoured and a jackscrew-operated, two-position, radiation cooled extendible nozzle is added. The primary nozzle exit diameter is fixed at 40 in., since this is the limiting diameter for the extendible nozzle to be retracted over the engine's power head and is also the largest size which allows installation with a truncated extendible nozzle in P&WA/GPD E-6 and E-7 test stands. The injector is reoptimized to give improved performance at a mixture ratio of 6.0. The engine maintains the same design margins as the RL10A-3-3 engine since the chamber pressure level remains unchanged and the turbopumps are basically unchanged.

The dry weight of the engine and its subassemblies are summarized in Table 2-2. Of the total engine weight of 431 lb, 35% consists of the weight of the existing hardware, 57% is calculated from layout drawings, and 8% is estimated.

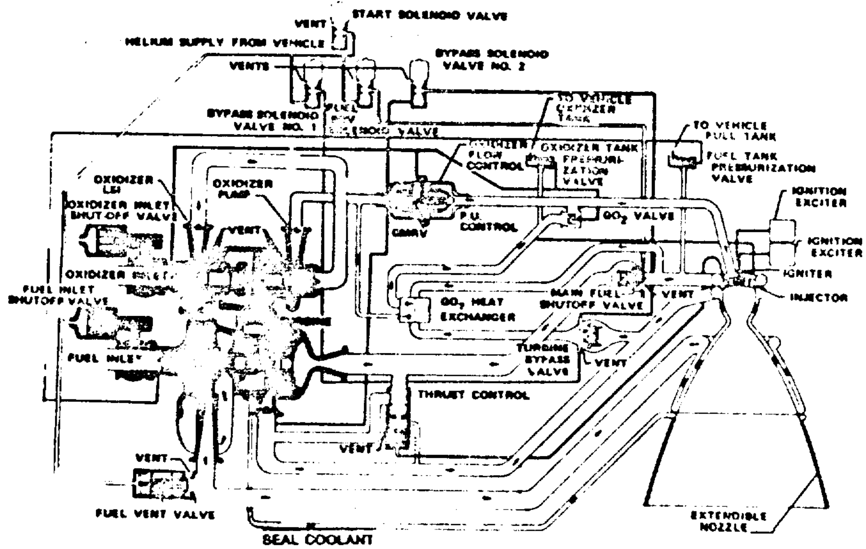


Figure 2-6. RL10 Derivative IIA — Propellant Flow Schematic

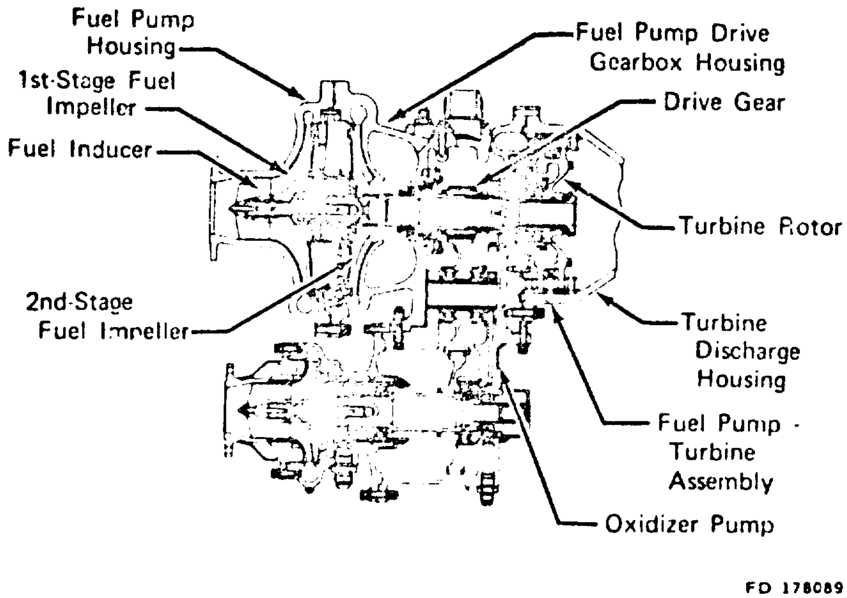


Figure 2-7a. RL10A-3-3 Turbopump Assembly Shown in Comparison With Derivative IIA Turbomachinery (Refer to Figure 2-7b)

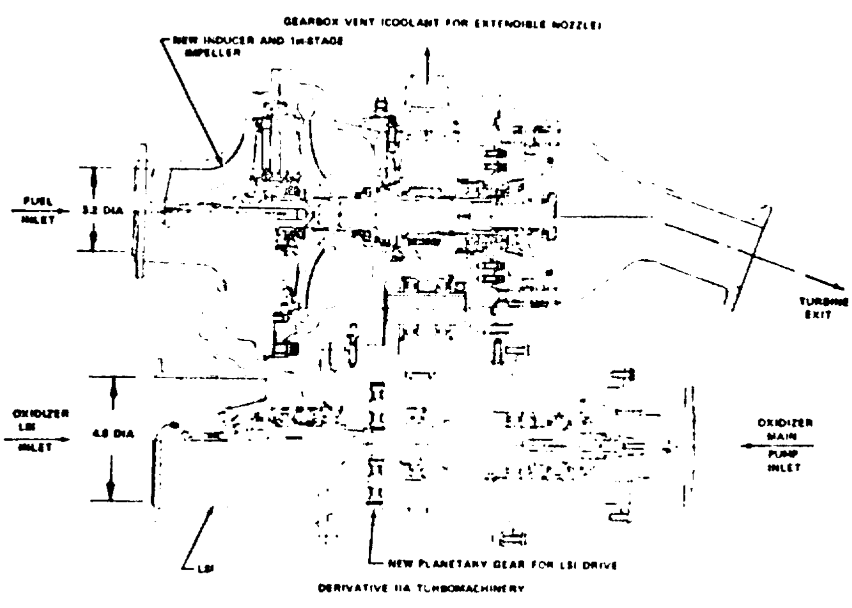


Figure 2-7b. Derivative IIA Turbopump Assembly Shown in Comparison With RL10A-3-3 Turbomachinery (Refer to Figure 2-7a)

TABLE 2-2. RL10 DERIVATIVE IIA ENGINE WEIGHT

Turbopumps and Gearbox	99 lb
Thrust Chamber and Primary Nozzle	110 lb
Extendible Nozzle Actuation System	40 lb
Extendible Nozzle	25 lb
GO_2 Heat Exchanger	13 lb
Controls, Valves & Actuators	83 lb
Plumbing and Miscellaneous Hardware	46 lb
Ignition System	15 lb
	431 lb

2.1.3 Operation and Performance Characteristics

2.1.3.1 Operation

The engine is started in THI mode, with propellants supplied in vapor, mixed, or liquid phases.

With the inlet shutoff valves open, fuel flows through the pump, the thrust chamber cooling jacket, around the turbine, through the GO_2 heat exchanger, and into the main injector. Similarly, the oxidizer flows through the pump, and with the oxidizer flow control valve shut, all the flow goes through the heat exchanger to the injector. The operating conditions shown on Figure 2-8 are for a thermally conditioned engine with liquid propellants supplied at 16 psia.

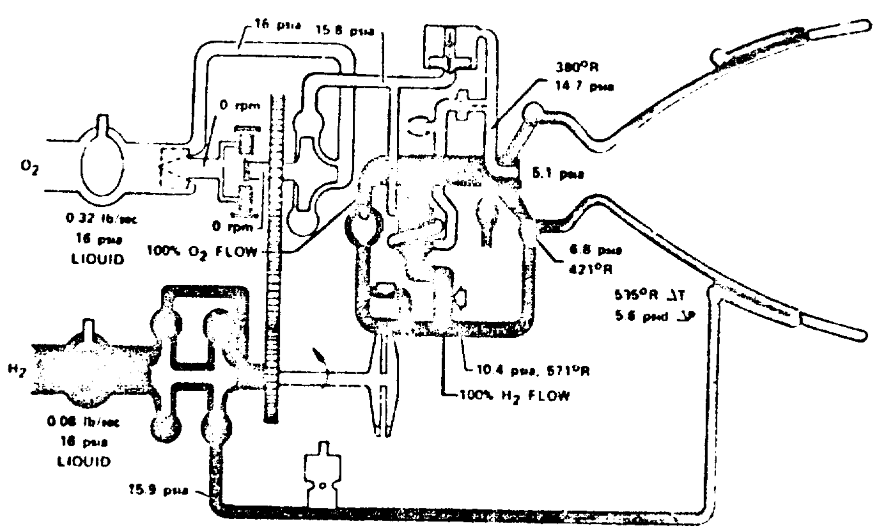


Figure 2-8. RL10 Derivative IIA Propellant Flow Schematic — Tank Head Idle Mode

After pump conditioning has been completed in THI mode, the engine is ready to be brought to its maneuver thrust level for low ΔV maneuvers or as a step on its acceleration to full thrust. To start the turbopumps, the main fuel shutoff valve is opened, and the turbine bypass valve is closed momentarily to give a high initial turbine torque and is then reopened to the maneuver thrust position. Figure 2-9 shows the engine operating at maneuver thrust.

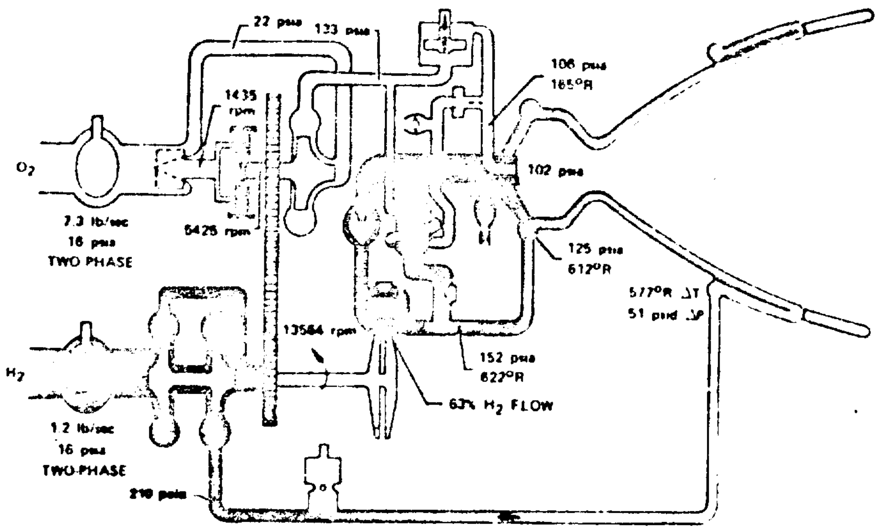


Figure 2-9. RL10 Derivative IIA Propellant Flow Schematic — Maneuver Thrust

By closing the turbine bypass valve, the engine is accelerated to full thrust. At about 90% of full thrust, the thrust control valve opens to reduce thrust overshoot. Operation of the engine a full thrust and 6.0 mixture ratio is shown in Figure 2-10.

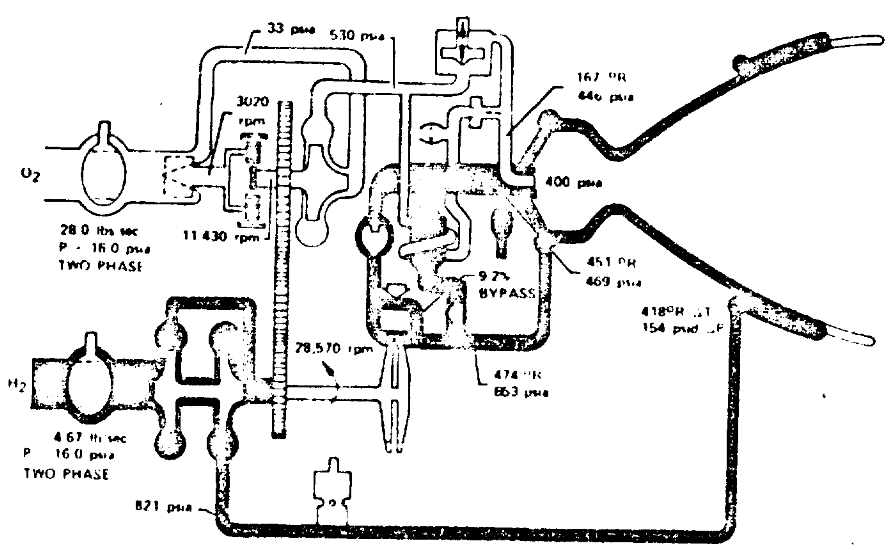


Figure 2-10. RL10 Derivative IIA Propellant Flow Schematic — Full Thrust (MR=6.0)

2.1.3.2 Performance

The steady-state performance characteristics of the RL10 Derivative IIA engine are summarized in Table 2-3.

TABLE 2-3. PERFORMANCE CHARACTERISTICS OF RL10 DERIVATIVE IIA ENGINE

Operating Mode	*Tank Head Idle*	*Maneuver Thrust*	*Full Thrust*
Thrust, lb	170	3,750	15,000
Mixture Ratio	4.0	6.0	6.0
Chamber Pressure, psia	5.1	102	400
Specific Impulse, sec	438	446	459.8
Fuel Turbopump Speed, rpm	0	13,564	28,570
Oxidizer and Fuel Pump Inlet Condition Limits	>16 psia (Superheated, mixed phase or liquid)	>10 psia, <70% Vapor	>10 psia, <40% Vapor

2.1.4 Programmatics

2.1.4.1 Engine Development

The total development program without predevelopment activity for the baseline Derivative IIA engine will require a 64-month design, fabrication, and test effort. This effort will encompass three design/build/test cycles to Final Flight Certification (FFC). Figure 2-11 shows the development schedule and presents the major program milestones and key decision points as well as the total development program. The total development funding requirements and cost breakdown by function are given in Figure 2-12.

The total and design verification development programs (including propellant costs) are summarized in Table 2-4.

2.1.4.2 Production Engine Cost

The estimated first unit cost of a Derivative IIA production engine is $1.56 million.

2.1.4.3 Operational and Flight Support Costs

Summary of Costs:

Missions per year	15	30	45
Total costs for 12 years, $ Million	78.2	86.0	102.7

TABLE 2-4. SUMMARY OF TOTAL AND DESIGN VERIFICATION PROGRAMS, RL10 DERIVATIVE IIA

	Total Development Program	–	*Design Verification Program*	=	*Allowance for Redesign and Reverification*
Equivalent Sets of Engines	26	–	10	=	16 (61.5%)
Engine Builds and Rebuilds	90	–	58	=	32 (35.5%)
Engine Tests	850	–	550	=	300 (35%)
Duration to FFC in Months	64	–	56	=	8 (12.5%)
Total Cost, $ Million	99.9	–	59.7	=	40.2 (40.2%)

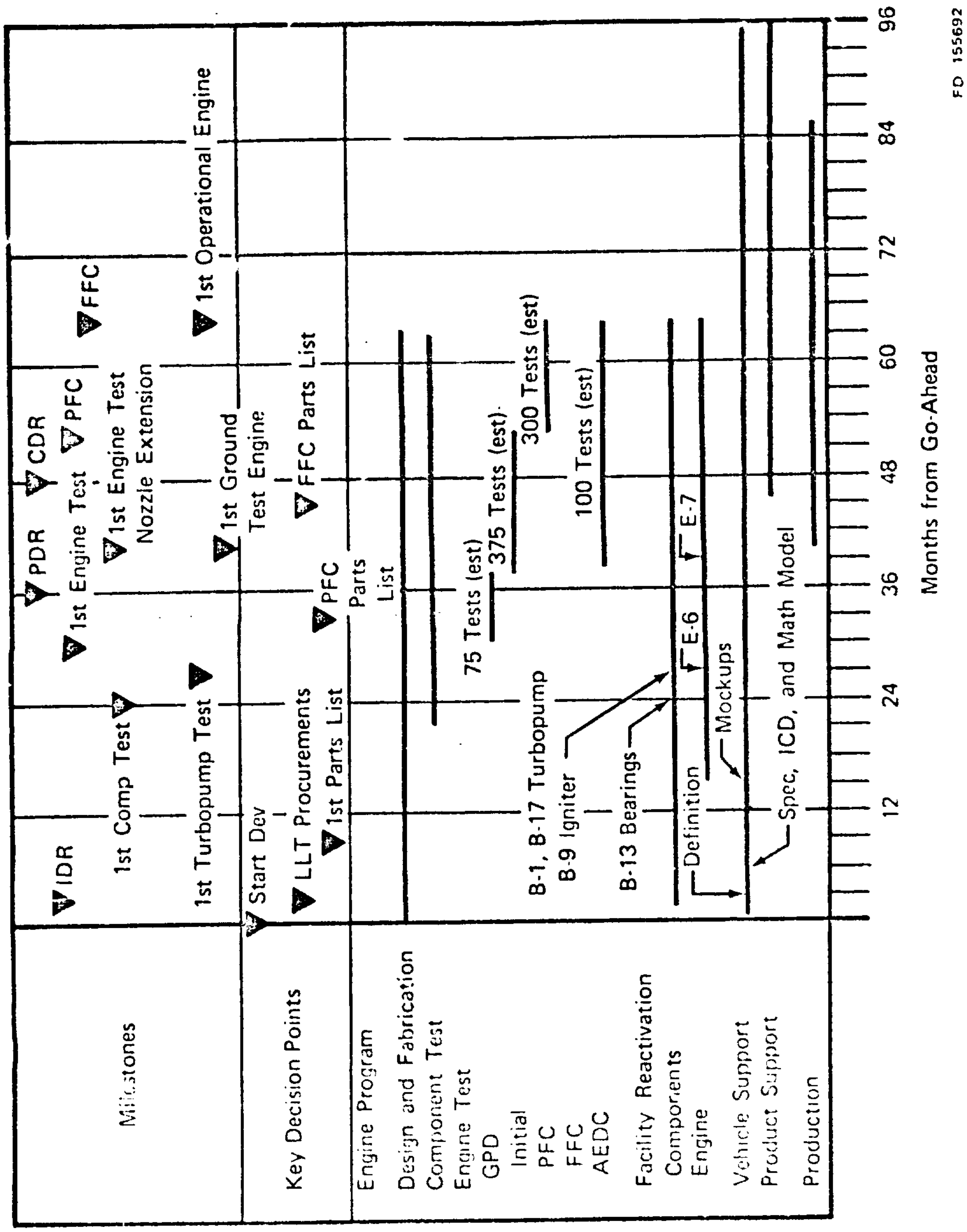


Figure 2-11. Derivative IIA Development Schedule and Major Program Milestones

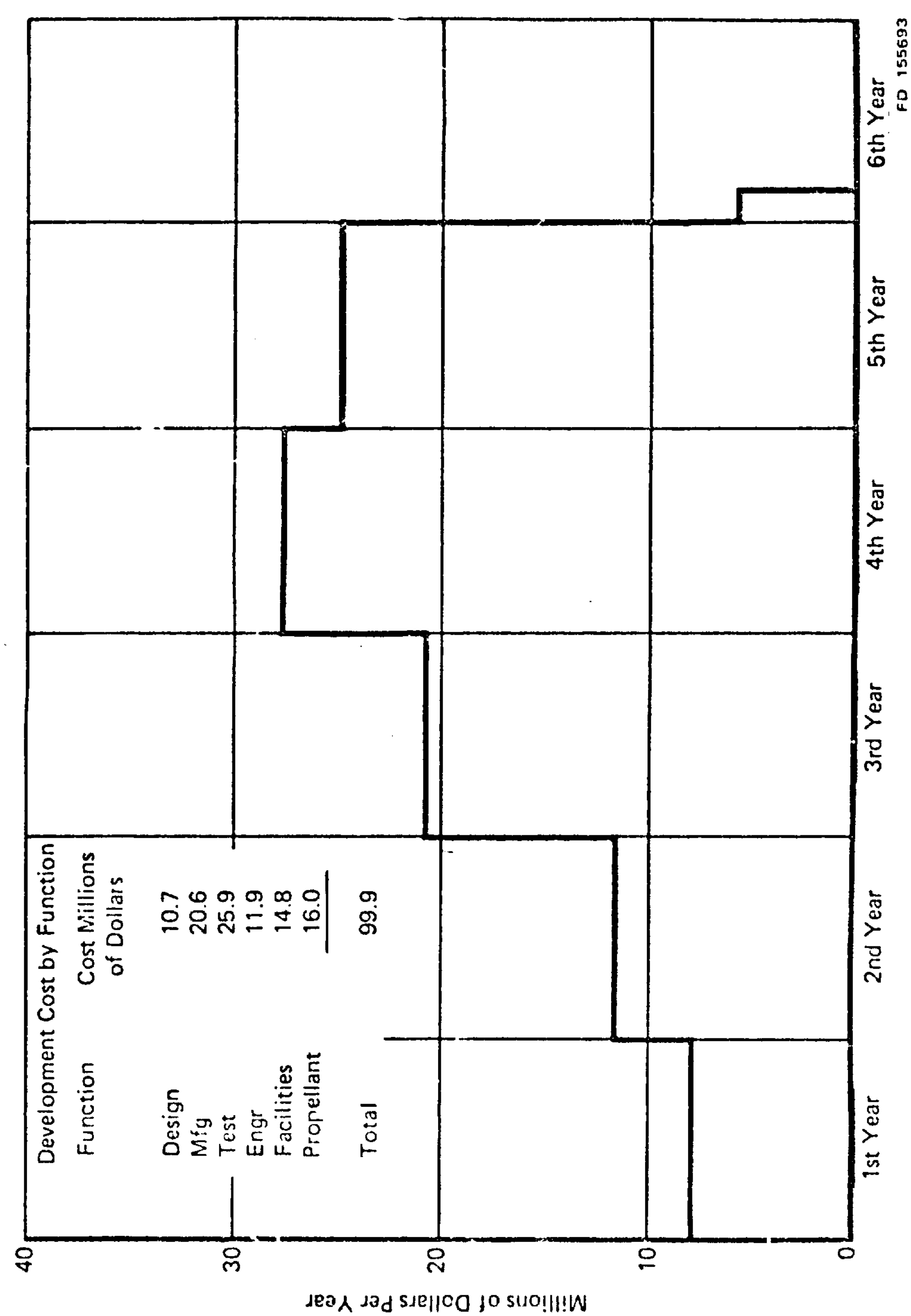
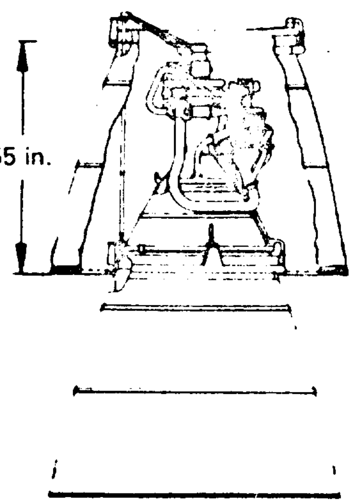


Figure 2-12. Derivative IIA Total Development Program Cost and Funding

2.2 RL10 DERIVATIVE IIB ENGINE



Thrust	: 15,000 lb
Chamber Pressure	: 400 psia
Area Ratio	: 205
I_{sp}	: 459.8 sec at 6.0 MR
Operation	: Full Thrust (Low NPSH)
	: Pumped Idle (Saturated Propellants)
Conditioning	: Tank Head Idle
Weight	: 392 lb
Life (TBO)	: 190 Firings/5 hr
DDT&E Cost	: $79 Million

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The RL10 Derivative IIB is similar to the Derivative IIA engine except that it does not have the requirement for two-phase pumping capability at full thrust. Therefore, this summary will emphasize those aspects of the Derivative IIB engine which differ from the Derivative IIA.

2.2.1 Definition and Requirements

The RL10 Derivative IIB is defined as the basic RL10A-3-3 engine with the following changes:

1. Two-position nozzle with recontoured primary section
2. Reoptimized injector
3. Tank head idle mode
4. Pumped idle mode, with saturated propellants in vehicle tanks, and bootstrap autogenous pressurization. This mode of operation allows the RL10A-3-3 Bill of Material turbopump to be run at a sufficiently low speed where prepressurization subcooling of the propellants at the pump inlets is not required. By using the engine's bootstrap autogenous pressurization capability, the tanks can then be prepressurized to satisfy the engine's full thrust pump inlet net positive suction head (NPSH) requirements before acceleration to full thrust.

2.2.2 Description

The general arrangement of the RL10 Derivative IIB engine is shown in the installation drawings in Figures 2-13 and 2-14. This engine is interchangeable with the RL10A-3-3.

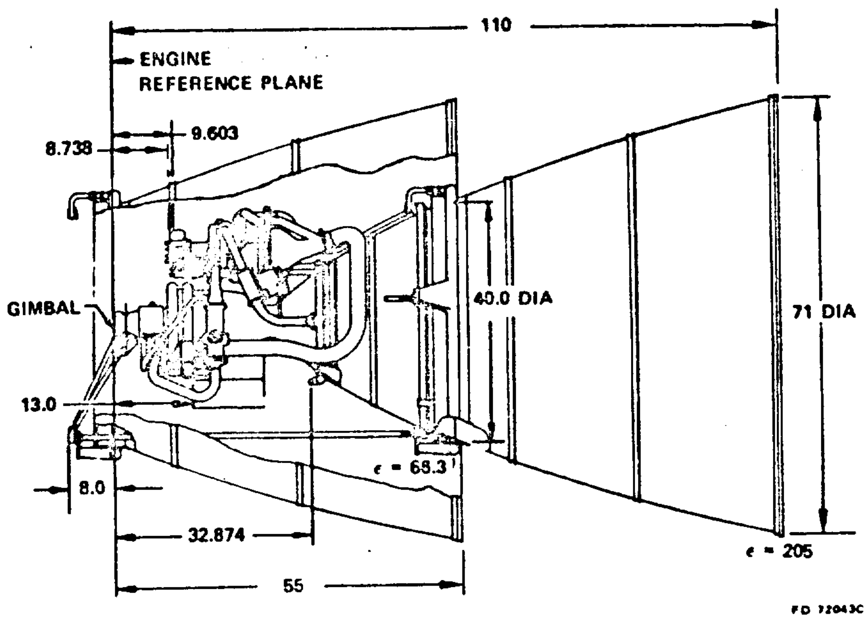


Figure 2-13. Derivative IIB Engine Installation Drawing (Sheet 1)

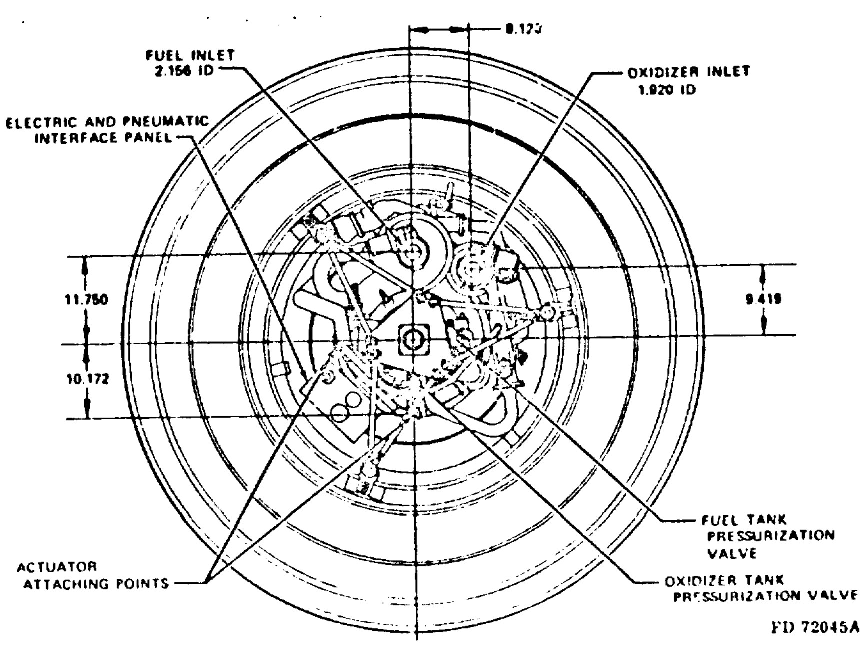


Figure 2-14. Derivative IIB Engine Installation Drawing (Sheet 2)

The principal components of the RL10 Derivative IIB engine are shown in Figure 2-15. This engine uses the RL10A-3-3 Bill-of-Material turbopump and inlet valves but in other respects is essentially the same as the Derivative IIA engine. Like the Derivative IIA engine, this engine also maintains the RL10A-3-3 engine design margins.

The dry weight of the engine and its subassemblies are summarized in Table 2-5. Of the total engine weight of 392 lb, 44% is weight of existing hardware, 43% is calculated from layout drawings, and 13% is estimated.

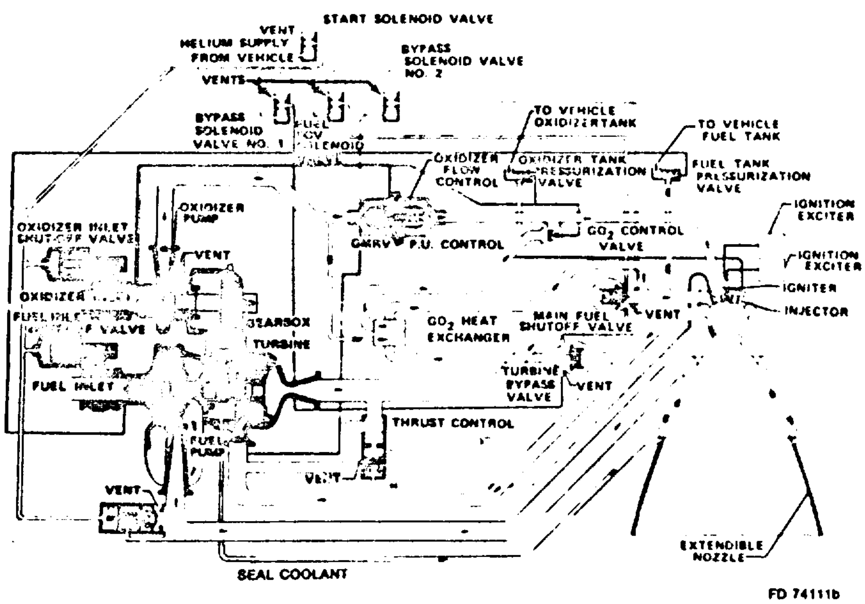


Figure 2-15. RL10 Derivative IIB Propellant Flow Schematic

TABLE 2-5. RL10 DERIVATIVE IIB ENGINE WEIGHT

Turbopump and Gearbox	79 lb
Thrust Chamber and Primary Nozzle	110 lb
Extendible Nozzle Actuator System	40 lb
Extendible Nozzle	25 lb
GO_2 Heat Exchanger	13 lb
Controls, Valves & Actuators	66 lb
Plumbing and Miscellaneous Hardware	44 lb
Ignition System	15 lb
Total Dry Weight	392 lb

2.2.3 Operation and Performance Characteristics

2.2.3.1 Operation

The operation of this engine is basically the same as that of the Derivative IIA, and propellant settling and engine thermal conditioning in THI is carried out in the same manner.

Further analysis of RL10 pump cavitation data carried out during the 1973 study showed that the Bill-of-Material inducers could be operated at well over 20% of full thrust without the propellants needing to be prepressurized. By increasing the thrust level for pumped idle mode to 25% of full thrust, the need for a cavitating venturi in the fuel system and the risk of oxidizer injector flow instability were both eliminated, while the prepressurization supply pressure was increased. The thrust level for the pumped idle mode of the Derivative IIB was therefore made the same as the maneuver thrust level of the Derivative IIA, and acceleration to this operating mode is carried out in the same way. A wide range of prepressurization flowrates can be supplied in pumped idle mode with little change in engine thrust, although active control valves are not used.

Acceleration to full thrust and operation at this level is the same as that of the Derivative IIA engine, except that propellants with positive NPSH have to be supplied to the engine, either by using the engine's autogenous pressurization system or with some vehicle-supplied system, i.e., boost pumps, helium pressurization, etc. Operation of the engine at full thrust and 6.0 mixture ratio is as shown in Figure 2-16.

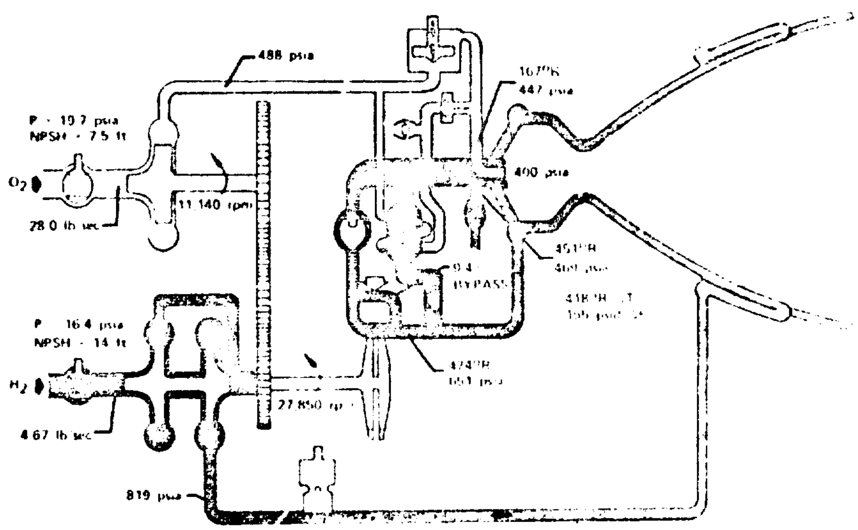


Figure 2-16. RL10 Derivative IIB Propellant Flow Schematic -- Full Thrust (MR = 6.0)

2.2.3.2 Performance

The steady-state performance characteristics of the RL10 Derivative IIB engine are summarized in Table 2-6.

TABLE 2-6. PERFORMANCE CHARACTERISTICS OF RL10 DERIVATIVE IIB ENGINE

Operating Mode	*Tank Head Idle*		*Maneuver Thrust*	*Full Thrust*
Thrust, lb	170		3,750	15,000
Mixture Ratio	4.0		6.0	6.0
Chamber Pressure, psia	5.2		102	400
Specific Impulse, sec	438		446	459.8
Fuel Turbopump Speed, rpm	0		13,360	27,850
Pump Inlet Condition Limits				
Fuel	>16 psia	Superheated Mixed phase or Liquid	>10 psia <65% Vapor	>14 ft NPSH
Oxidizer	>16 psia		>10 psia <45% Vapor	>7.5 ft NPSH
Fuel System Pressurization Supply	NA		0 to 0.15 lb/sec 550° to 520°R	0 to 0.1 lb/sec 440°R
Oxidizer System Pressurization Supply	NA		0 to 2 lb/sec 570° to 230°R	0 to 1 lb/sec 450° to 255°R

2.2.4 Programmatics

2.2.4.1 Engine Development

The total development program without predevelopment activity for the baseline Derivative IIB engine will require 58 months of design, fabrication, and test effort. This effort will encompass three design/build/test cycles to FFC. Figure 2-17 shows the development schedule and presents the major program milestones and key decision points. The total development funding requirements and cost breakdown by function are given in Figure 2-18.

The total and design verification development programs (including propellant costs) are summarized in Table 2-7.

2.2.4.2 Production Engine Cost

The estimated first unit cost of a Derivative IIB engine is $1.50 million.

2.2.4.3 Operation Flight Support Costs

Summary of Costs:

Missions per year	15	30	45
Total costs for 12 years, $ Million	68.9	76.5	93.0

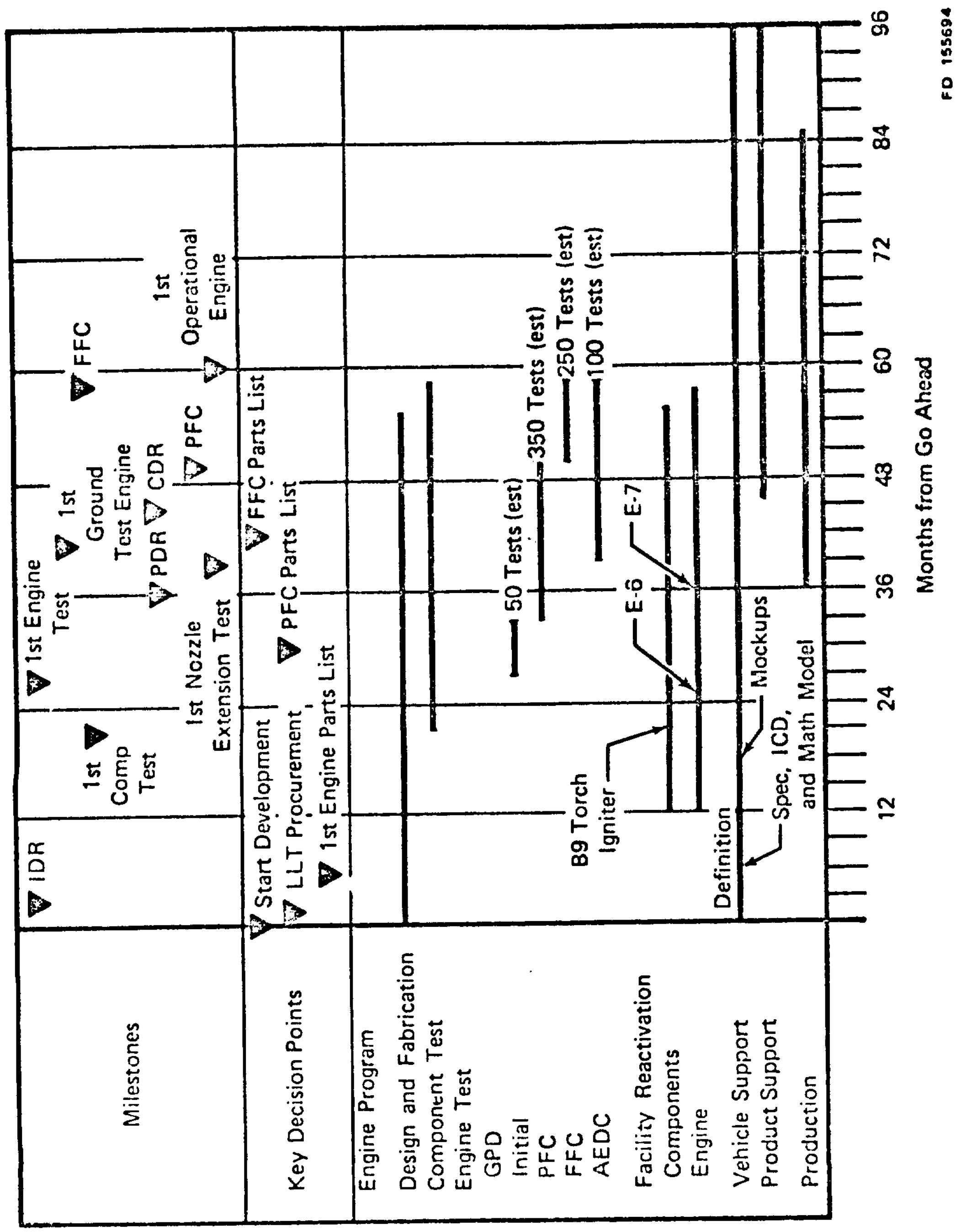


Figure 2-17. Derivative IIB Development Schedule and Major Program Milestones

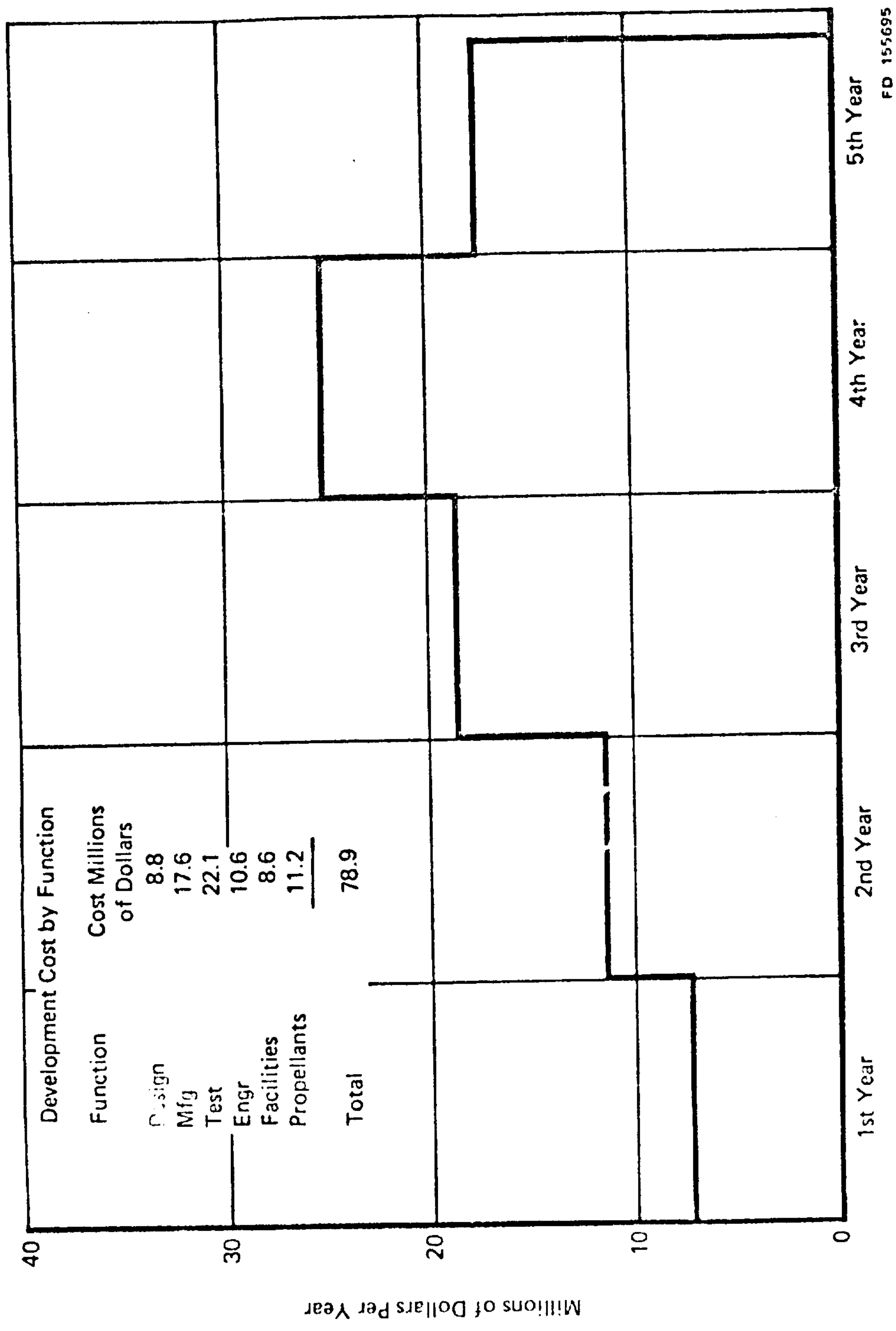
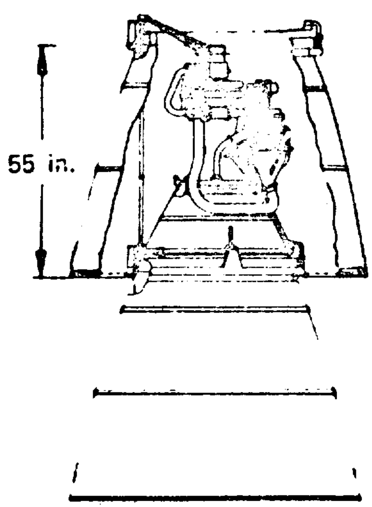


Figure 2-18. Derivative IIB Total Development Program Cost and Funding

TABLE 2-7. SUMMARY OF TOTAL AND DESIGN VERIFICATION PROGRAMS, RL10 DERIVATIVE IIB

	Total Development Program	–	*Design Verification Program*	=	*Allowance for Redesign and Reverification*
Equivalent Sets of Engines	23	–	9	=	14 (61%)
Engine Builds and Rebuilds	80	–	53	=	27 (34%)
Engine Tests	750	–	500	=	250 (33%)
Duration to FFC in Months	59	–	50	=	9 (15%)
Total Cost $ Million	78.9	–	46.6	=	32.3 (41%)

2.3 RL10 DERIVATIVE IIC ENGINE



Thrust	: 15,000 lb
Chamber Pressure	: 400 psia
Area Ratio	: 205
I_{sp}	: 458.6 sec at 6.0 MR
Operation	: Full Thrust (Low NPSH)
Conditioning	: Overboard Dump
Weight	: 374 lb
Life (Expendable Mission)	: 10 Firings/1.25 hr
DDT&E Cost	: $21 Million

FD 75662a

The RL10 Derivative IIC engine is included in this report, even though it was not one of the engines defined in the original study, because it is a low-cost, high-performance candidate engine for an early expendable OTV.

2.3.1 Definition and Requirements

The RL10 Derivative IIC is the existing RL10A-3-3 engine, with the addition of a high-area-ratio, two-position nozzle and requalified to operate under OTV conditions. As a result, there are the following changes in engine requirements from those of the RL10A-3-3 Bill-of-Material engine:

a. Two position nozzle with recontoured primary section

b. Mixture ratio increased to 6.0 (± 0.5)

c. H_2 autogenous pressurization

d. Increased life

e. 50% reduced NPSH limit and minimum pump inlet pressures reduced from 50 to 28 psia (H_2) and from 45 to 35 psia (O_2).

2.3.2 Description

The general arrangement of the RL10 Derivative IIC engine is shown by the engine installation drawings in Figures 2-19 and 2-20.

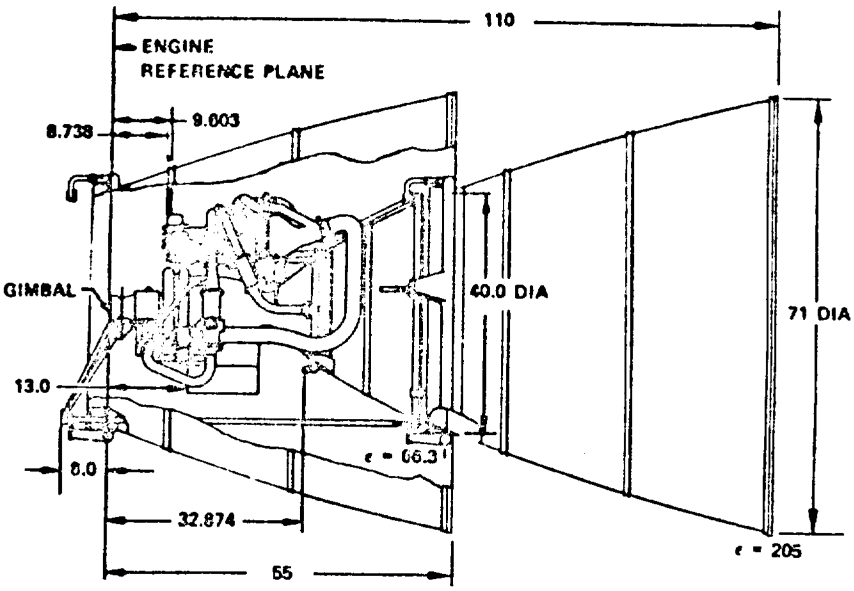


Figure 2-19. RL10 Derivative IIC Installation Drawing (Sheet 1)

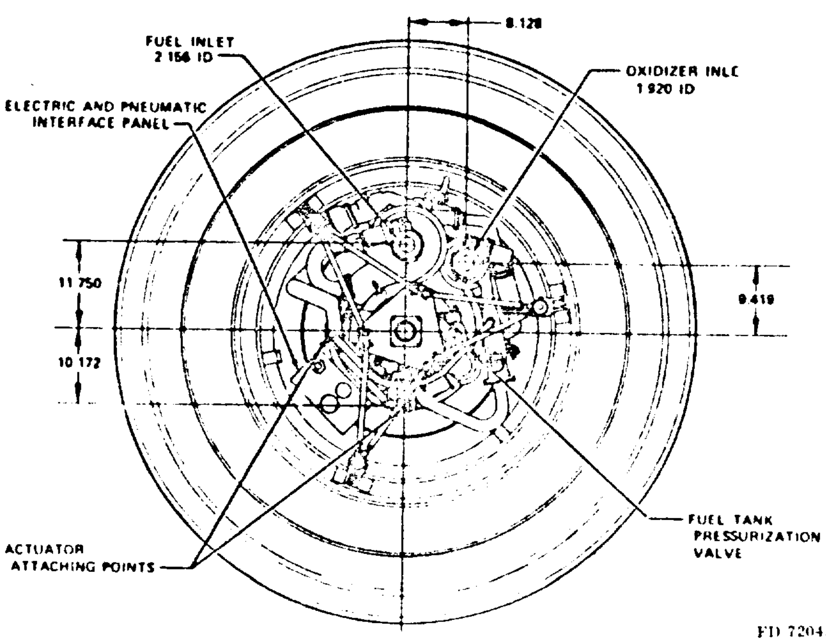


Figure 2-20. RL10 Derivative IIC Installation Drawing (Sheet 2)

The principal components of this engine are depicted in the flow schematic in Figure 2-21. Other than the two-position nozzle and actuation system, only minor modifications from the Bill-of-Material RL10A-3-3 are required. An extra solenoid valve is added to minimize overboard cooldown propellants needed to prestart condition the engine. The basic RL10 has been satisfactorily run at mixture ratios well in excess of 7:1; thus, only minor trim adjustments are required to meet the 6.0 mixture ratio requirement. The fuel tank pressurization valve is the same design that was used on the RL10-powered S-IV vehicle. No modifications are required to meet the increased life requirement, since the major RL10 engine components have a demonstrated life of greater than 190 firings and 5 hr. Similarly, no modifications are required to meet the reduced NPSH and minimum pump inlet pressure requirements.

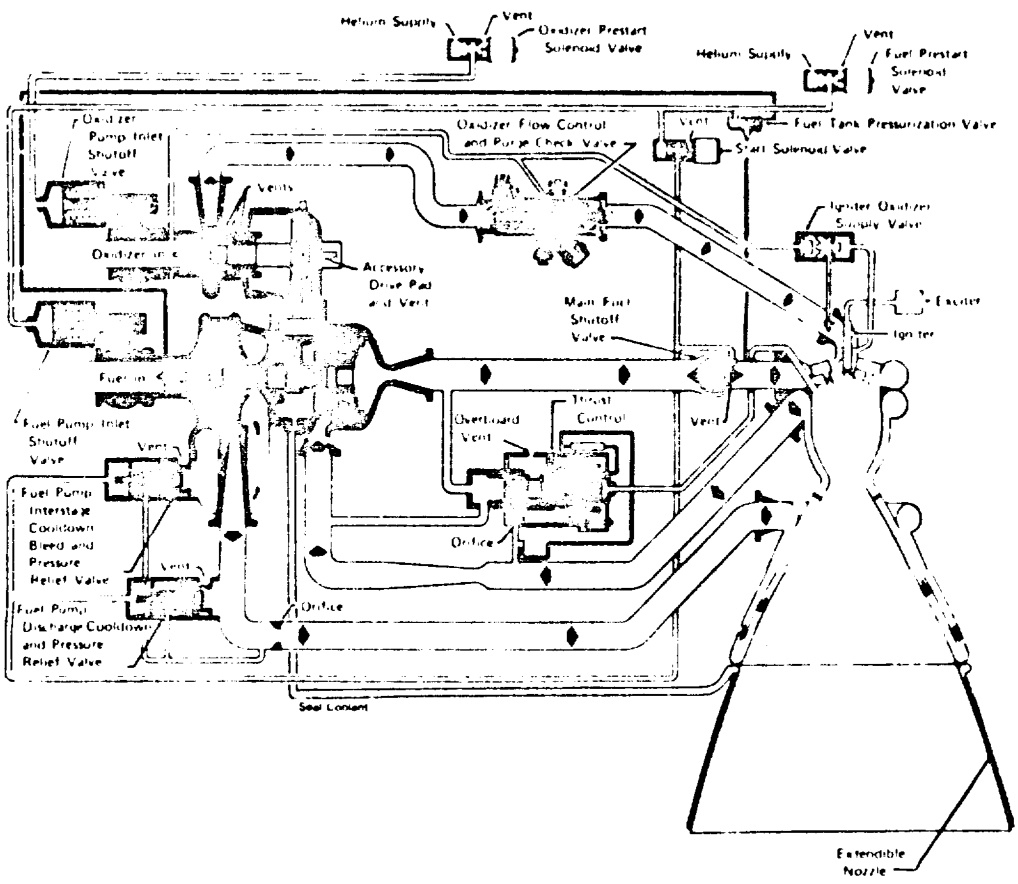


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Figure 2-21. Propellant Flow Schematic for Derivative IIC Engine

The dry weight of the engine and its subassemblies are summarized in Table 2-8. Of the total engine weight of 374 lb, 50% is weight of existing hardware, 35% is calculated from layout drawings, and 15% is estimated.

TABLE 2-8. RL10 DERIVATIVE IIC ENGINE WEIGHT

Turbopump and Gearbox	79 lb
Thrust Chamber and Primary Nozzle	110 lb
Extendible Nozzle Actuator System	40 lb
Extendible Nozzle	25 lb
Controls, Valves and Actuators	61 lb
Plumbing and Miscellaneous Hardware	44 lb
Ignition System	15 lb
Total Dry Weight	374 lb

2.3.3 Operation and Performance Characteristics

2.3.3.1 Operation

This engine is conditioned, like the RL10A-3-3, by dumping propellants through the engine before the turbopump is started. By adding the third solenoid, which allows the optimization of fuel and oxidizer chilldown times, the amount of propellants needed to condition the engine are reduced. The fuel system is thermally conditioned by hydrogen which enters through the inlet shutoff valve and then passes through both stages of the unshrouded fuel pump. With the main fuel shutoff valve closed, this hydrogen is vented overboard through the interstage and discharge cooldown valves. The oxidizer system is thermally conditioned by oxygen entering through the inlet shutoff valve, moving through the pump, and with flow restricted by the small starting flow area of the oxidizer control valve, the oxygen is discharged overboard through the oxidizer nozzles in the injector and out of the thrust chamber.

The engine is started by energizing the igniter and opening the main fuel shutoff valve. The fuel proceeds through the pass and a half-tubular thrust chamber where it cools the chamber and picks up the energy needed to drive the two stage turbine which powers the main pumps. The flow is controlled by bypassing the fuel around the turbine with the controller sensing thrust chamber pressure. The single-stage shrouded oxidizer pump and inducer is gear driven from the fuel turbopump. The engine fuel mixture ratio is varied by the oxidizer flow control valve which is controlled by the vehicle's propellant utilization system. Operation of the engine at full thrust and 6.0 mixture ratio is shown in Figure 2-22.

2.3.3.2 Performance

The present specification limits for the RL10A-3-3 minimum pump inlet pressure and NPSH requirements are in excess of the engine's demonstrated capability. These limits for the RL10 Derivative IIC engine have therefore been reduced, as shown in Table 2-9. This table also compares the main characteristics of the RL10 Derivative IIC with those of the Bill of Material RL10A-3-3 presently in production.

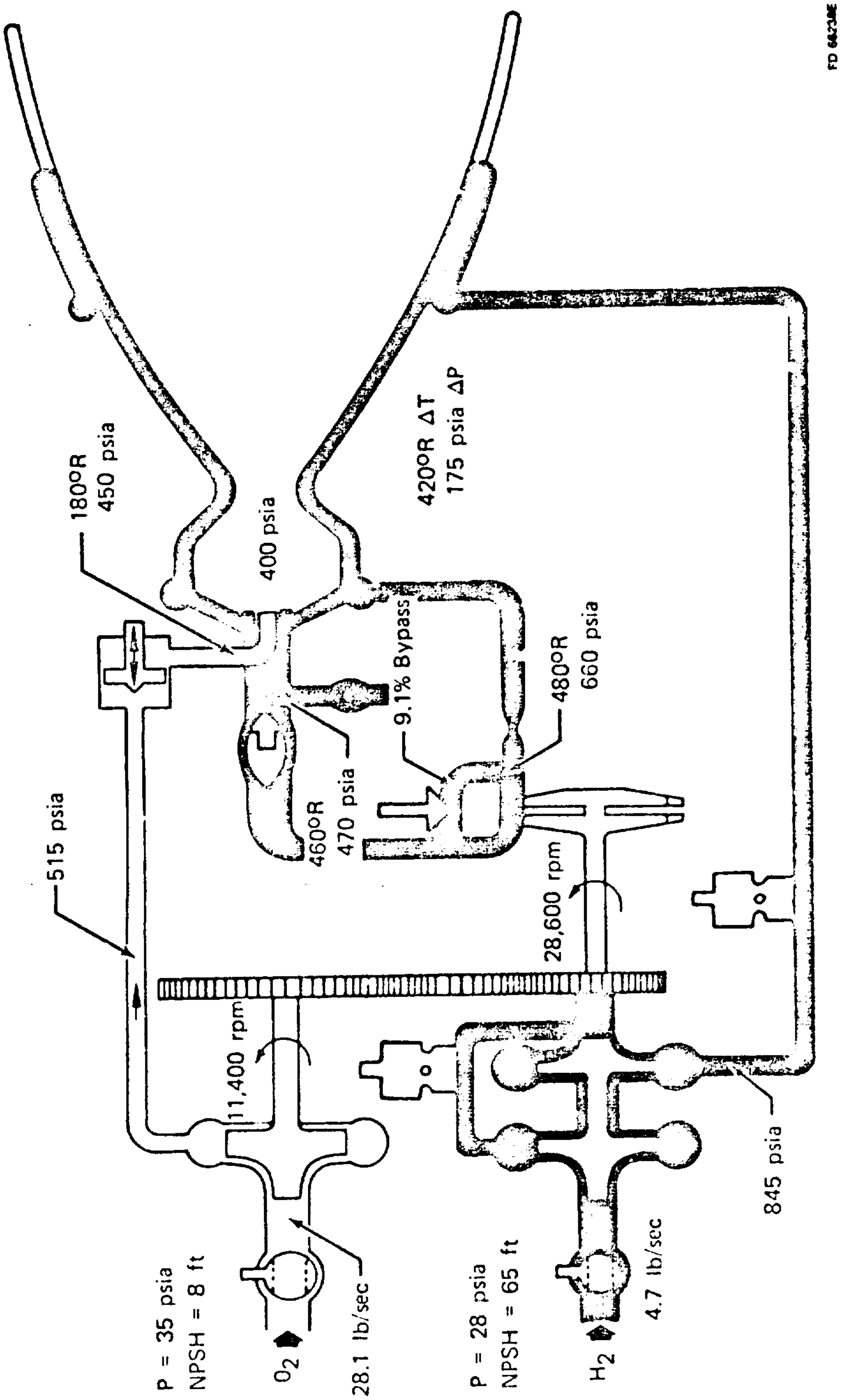

Figure 2-22. RL10 Derivative IIC Propellant Flow Schematic — Full Thrust (MR = 6.0)

TABLE 2-9. COMPARISON OF RL10 DERIVATIVE IIC AND RL10A-3-3 CHARACTERISTICS

	RL10 Derivative IIC	*RL10A-3-3*
Thrust, lb	15,000	15,000
Mixture Ratio	6.0	5.0
Chamber Pressure, psia	400	400
Specific Impulse, sec	458.6	444
Nozzle Area Ratio	205:1	57:1
Installed Length, in.	55	70
Dry Weight, lb	374	290*
Mission Life, firings/hours	>10/1.25	>3/1.1
H_2 Tank Pressurization Capability	Yes	No
Minimum Pump Inlet Pressure At Start — H_2/O_2, psia	28/35	30/45
Minimum Required NPSH — H_2/O_2, ft	65/8	135/16

*Excludes weight of instrumentation.

2.3.4 Programmatics

2.3.4.1 Engine Development

The total development program without predevelopment activity for the Derivative IIC engine requires 37 mo of design, fabrication, and test effort. This effort will encompass one design/build/test cycle to FFC. Figure 2-23 shows the total development program schedule and presents the major program milestones and key decision points. The total development funding requirements and cost breakdown by function are given in Figure 2-24.

Design Verification Specifications were not generated for this engine. The total development program is summarized as follows:

Equivalent Sets of Engines	8
Engine Builds and Rebuilds	10
Engine Tests	185
Total Cost — $ Million	20.6

If an increased life version of the RL10 Derivative IIC engine (in excess of the 10 firings/1.25 hr version) were required, it is estimated that only a modest addition to the program would be necessary to qualify the engine for a 60 firings/2.5 hr life. This addition would consist of 2 equivalent engine hardware sets, 1 additional engine rebuild, 60 engine tests and approximately $2 million dollars. This program would not, however, qualify the nozzle extension for the extended life; the life of this component would be determined in service by continuing use with frequent inspections until a cause for removal is found. It is expected, however, that the relatively favorable environment of the nozzle extension will allow the mission life requirements to be easily met or exceeded.

2.3.4.2 Production Engine Cost

The estimated first unit cost of a Derivative IIC production engine is $1.35 million.

2.3.4.3 Operation and Flight Support Costs

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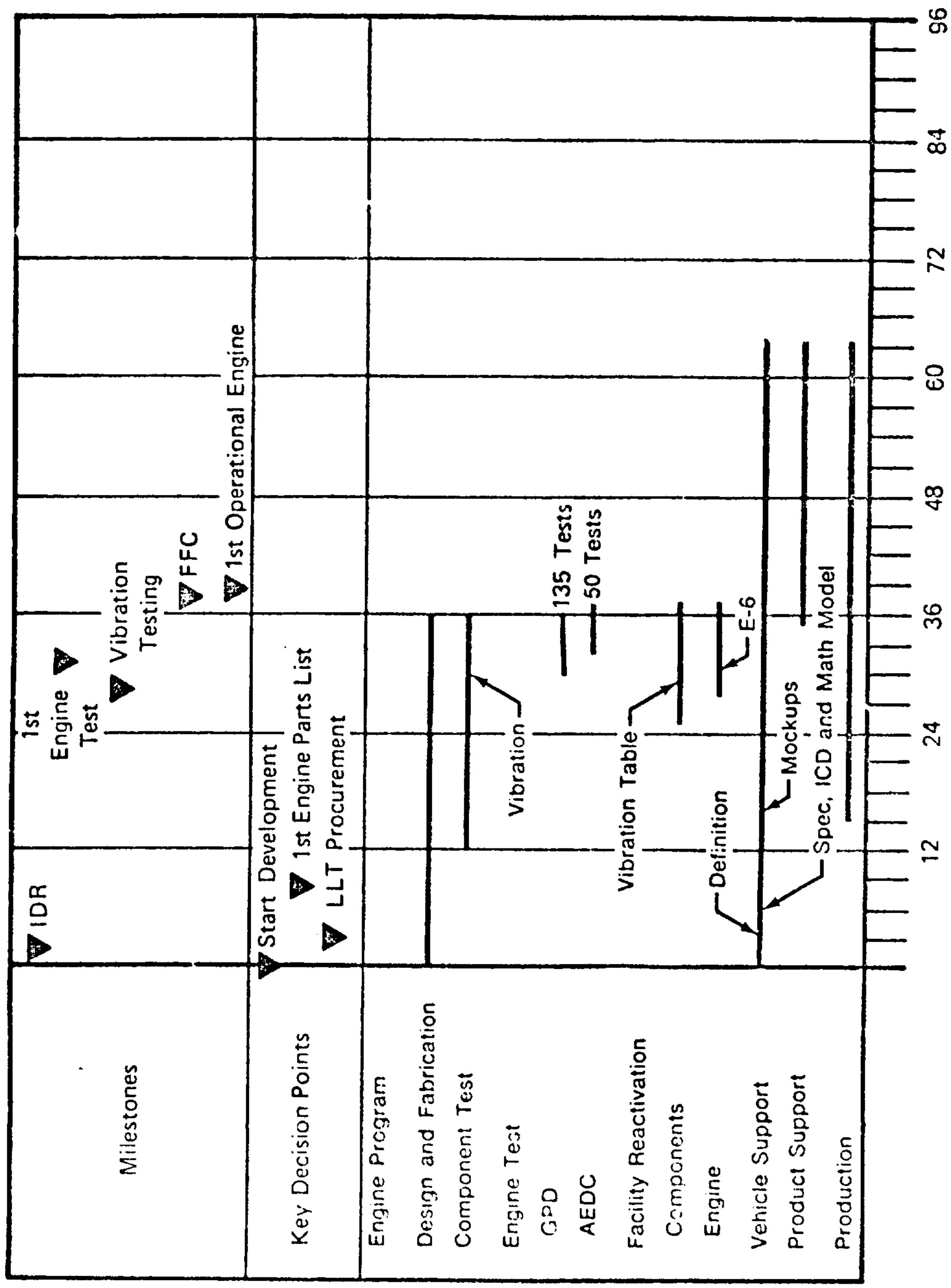


Figure 2-23. RL10 Derivative IIC Total Development Program Schedule and Major Milestones

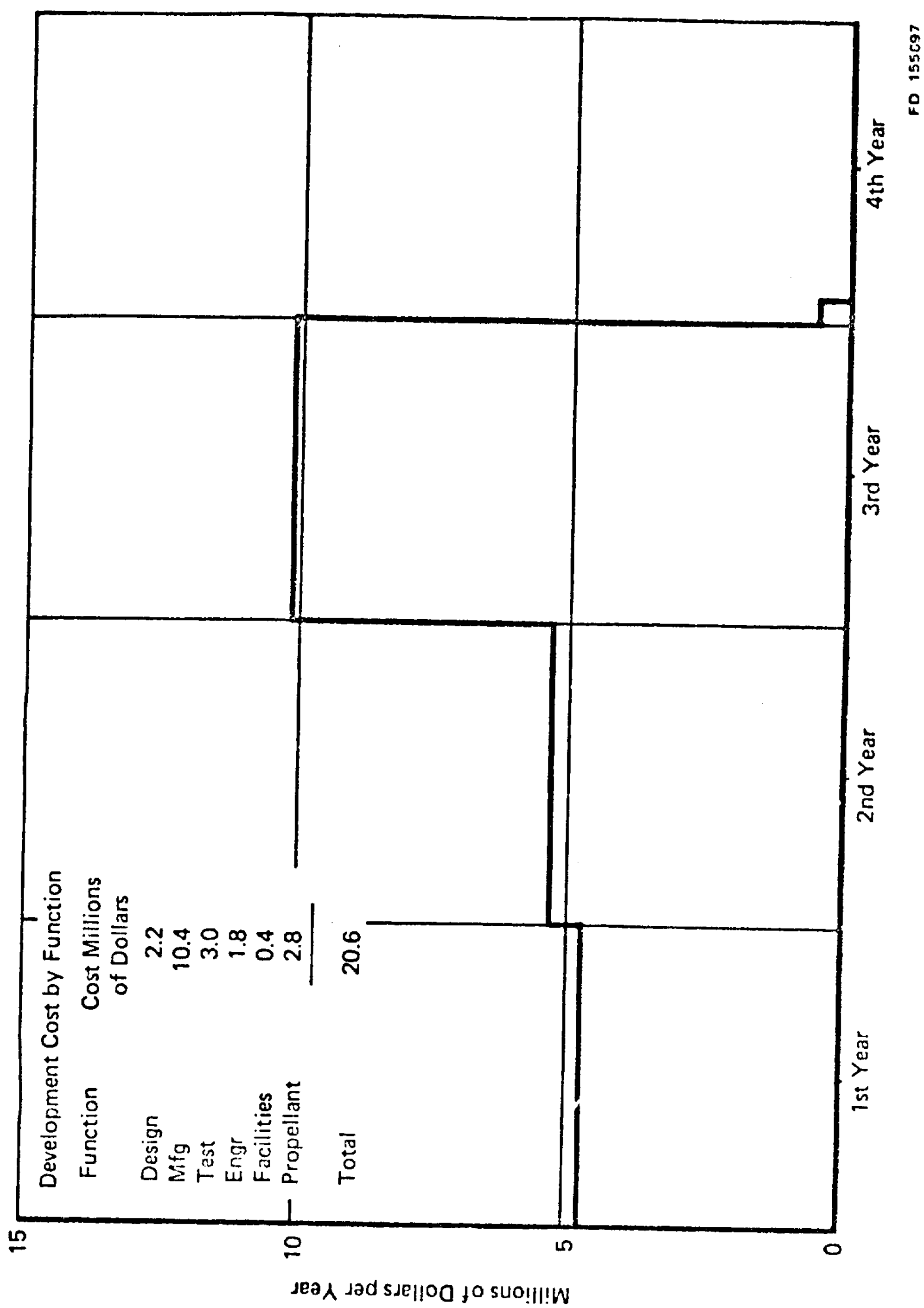
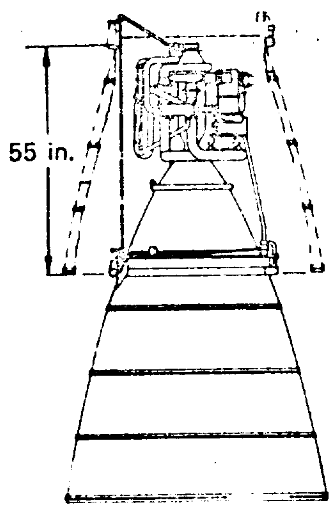


Figure 2-21. Derivative HC Development Program Cost and Funding

2.4 RL10 CATEGORY IV ENGINE



Thrust	: 15,000 lb
Chamber Pressure	: 915 psia
Area Ratio	: 388
I_{sp}	: 471.7 sec at 6.0 MR
Operation	: Full Thrust (Saturated Propellants)
	: Maneuver Thrust (Saturated Propellants)
Conditioning	: Tank Head Idle
Weight	: 371 lb
Life (Design TBO)	: 300 Firings/10 hr
DDT&E Cost	: $157 Million

FD 74124B

2.4.1 Definitions and Requirements

Unlike the Derivative II baseline engines, which are modified versions of the RL10A-3-3, the RL10 Category IV engine is a "clean sheet" design. However, it is not an advanced-technology engine, since it uses the same expander power cycle and basic design concepts of the RL10. Basically, it is a 1973 update of a design optimized specifically for use in the OTV. The baseline RL10 Category IV engine has the following requirements:

1. Interface requirements: interchangeable with RL10 Derivative IIA.
2. Operating modes: Same as RL10 Derivative IIA, i.e.,
   - Tank head idle mode
   - Maneuver thrust
   - Two-phase pumping capability at full thrust
3. Design life: 300 firings and 10 hr
4. Thrust level: 15,000 lb at 6.0 mixture ratio
5. Performance: optimize.

2.4.2 Description

The general arrangement of the RL10 Category IV engine is shown in the installation drawings on Figures 2-25 and 2-26. As required, this engine is interchangeable with the RL10 Derivative IIA, and the installed length was selected so that it would be the same as the minimum length Derivative IIA engine.

The principal components of the RL10 Category IV engine are shown in Figure 2-27. To obtain two-phase pumping capability at full thrust, fuel and oxidizer low-speed inducers are used. These inducers, which are of the same suction design as the oxidizer low-speed inducer on the RL10 Derivative IIA, are gear driven from high-speed pumps. As on the current RL10, a two-stage fuel pump is used with a two-stage turbine. The fuel impellers are fully shrouded to obtain high efficiency, since high tip speeds are not required. The single-stage oxidizer pump is driven by a single-stage turbine used in series with the fuel turbine. This was done so that the gears could be retained but the power transmitted through them could be reduced, leaving their stresses lower than in time RL10A-3-3. By gearing the pumps and inducers together, a complex closed-loop control system (and thus control problems especially during transient operation) is avoided.

(SHEET 1)

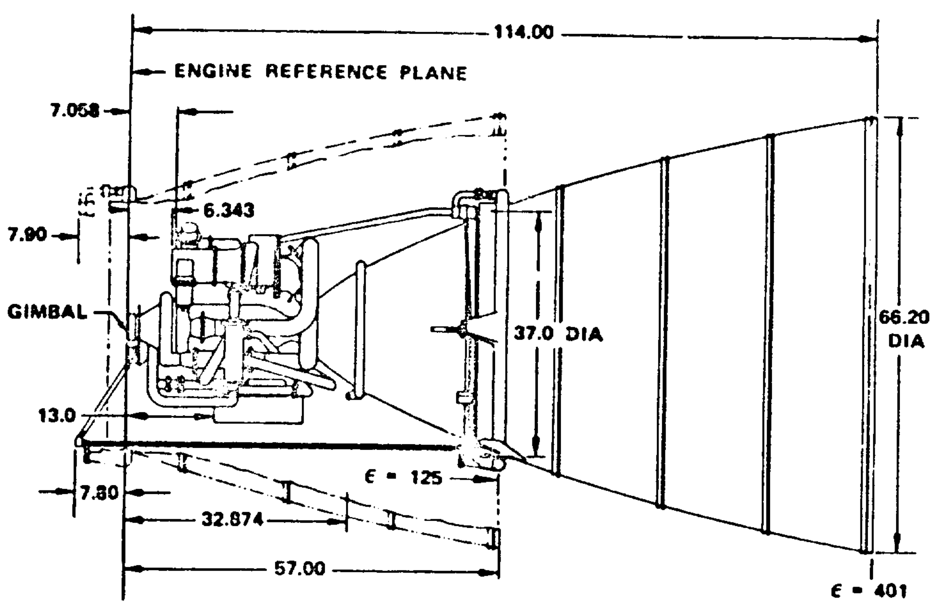


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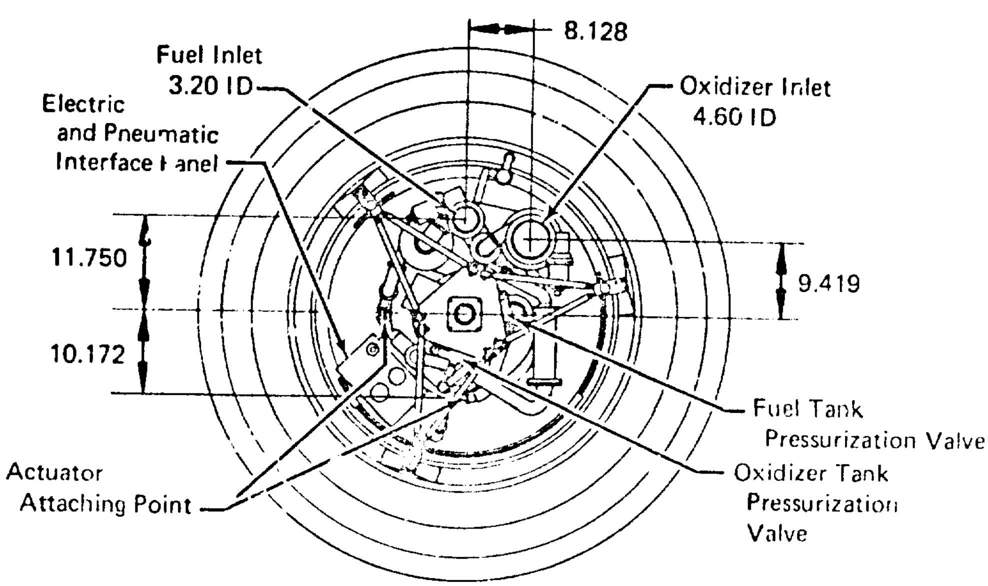
Figure 2-25. Category IV Engine Installation Drawing



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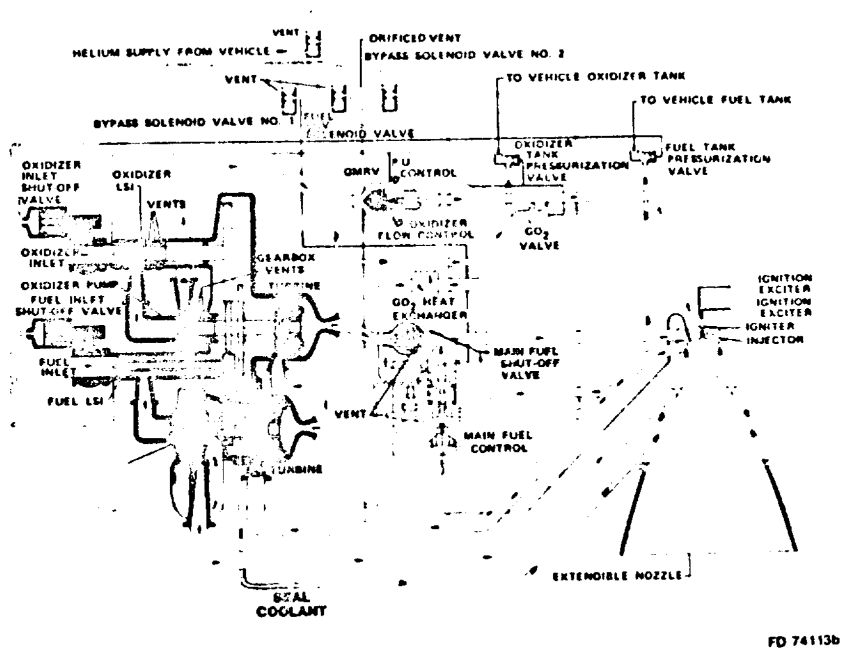


Figure 2-27. Category IV Propellant Flow Schematic

The combination of nickel alloy turbines and the expander power cycle give turbine life margin. To meet the thrust chamber life requirements, a milled channel chamber has been selected, but the design of the two-position extendable nozzle and primary nozzle is conceptually the same as that used in the Derivative II engines. The control system is also basically the same as that of the Derivative II engines, except that the main fuel control valve combines the functions of the thrust controller, turbine bypass, and fuel vent valves. The GO_2 heat exchanger design is the same as that used on the Derivative II engines. The igniter system utilizes redundant exciters and is of an updated lightweight design.

The dry weight of the engine and its subassemblies are summarized in Table 2-10. Of the total engine weight of 371 lb, 56% is calculated from layout drawings and 44% is estimated.

Table 2-10. RL10 CATEGORY IV ENGINE WEIGHT

Turbopumps and Gearbox	91 lb
Thrust Chamber and Primary Nozzle	87 lb
Extendible Nozzle Actuation System	35 lb
Extendible Nozzle	26 lb
GO_2 Heat Exchanger	13 lb
Controls, Valves and Actuators	74 lb
Plumbing and Miscellaneous Hardware	38 lb
Ignition System	7 lb
	371 lb

2.4.3 Operation and Performance Characteristics

2.4.3.1 Operation

The operation of this engine is basically the same as that of the Derivative IIA engine. With the start solenoid and bypass solenoid No. 1 energized, thermal conditioning in tank head idle mode is carried out with the engine operating as a pressure-fed system without turbopumps rotating. By de-energizing bypass solenoid No. 1 and energizing the other two solenoids, the main fuel shutoff valve is opened and the turbine bypass in the main fuel control valve is first closed, diverting all the fuel through the turbines, and subsequently, as turbine inlet pressure builds up, reopened to allow the engine to stabilize at maneuver thrust level. De-energizing bypass solenoid No. 2 closes this bypass accelerating the engine to full thrust, at which time the thrust controller element of the main fuel control valve takes over thrust control by regulating turbine bypass flow. Operation of the engine at full thrust and 6.0 mixture ratio is shown in Figure 2-28.

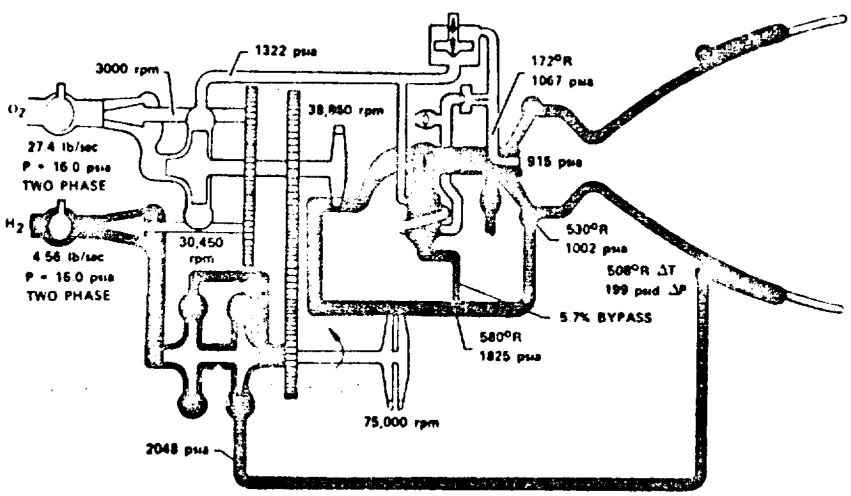


Figure 2-28. RL10 Category IV Propellant Flow Schematic — Full Thrust (MR = 6.0)

2.4.3.2 Engine Characteristics

The steady-state performance characteristics of the baseline RL10 Category IV engine are summarized in Table 2-11.

2.4.4 Programmatics

2.4.4.1 Engine Development

The total development program for the baseline Category IV engine requires 80 months of design, fabrication and test effort. This effort will encompass three design/build/test cycles to FFC. Figure 2-29 shows the total development program schedule and presents the major program milestones and key decision points. The total development funding requirements and cost breakdown by function are given in Figure 2-30.

Engine development is planned to be accomplished in the same manner as previously described for the Derivative II engines. Although design verification specifications (DVS's) were not formulated for the Category IV engine design during this study, an estimate of verification program requirements was made by comparing the Category IV engine design with the Derivative IIA engine design and deriving the design verification program for the Category IV engine from that of the Derivative IIA engine.

This derived design verification program is compared with the total development program in Table 2-12.

2.4.4.2 Production Engine Cost

The estimated first unit cost of a Category IV engine is $1.76 million.

2.4.4.3 Operation and Flight Support Costs

Summary of Costs:

Missions Per Year	15	30	45
Total Costs for 12 Years $ Million	86.1	94.5	103.5

TABLE 2-11. STEADY-STATE PERFORMANCE CHARACTERISTICS OF RL10 CATEGORY IV ENGINE

Operating Mode	*Tank Head Idle*	*Maneuver Thrust*	*Full Thrust*
Thrust, lb	79	3,750	15,000
Mixture Ratio	4.0	6.0	6.0
Chamber Pressure, psia	5.6	234	915
Specific Impulse, sec	445	458	471.7
Fuel Turbopump Speed, rpm	0	32,000	75,000
Fuel/Oxidizer Pump Inlet Condition Limits	≥16 psia Superheated Phase or Liquid	≥10 psia <70% Vapor	≥10 psia <40% Vapor

TABLE 2-12. COMPARISON OF DESIGN VERIFICATION PROGRAM WITH THE TOTAL DEVELOPMENT PROGRAM, RL10 CATEGORY IV

	Total Development Program	–	*Design Verification Program*	=	*Allowance for Redesign and Verification*
Equivalent Sets of Engines	30	–	11	=	19 (63%)
Engine Builds and Rebuilds	140	–	84	=	56 (40%)
Engine Tests	1,000	–	600	=	400 (40%)
Duration to FFC in Months	80	–	69	=	11 (14%)
Total Cost, $ Million	156.6	–	83.0	=	73.6 (47%)

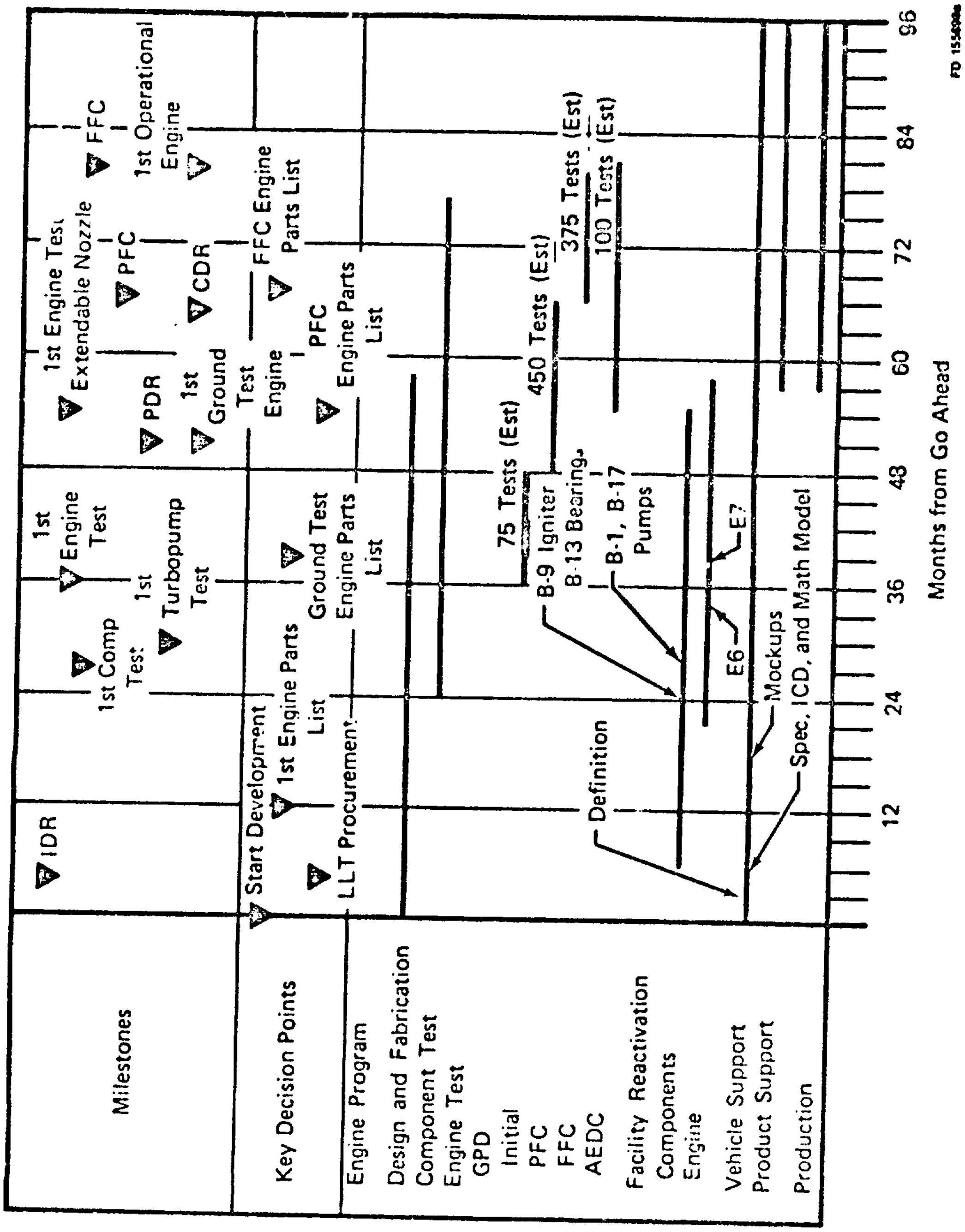


Figure 2-29. Category IV Program Schedule and Major Program Milestones

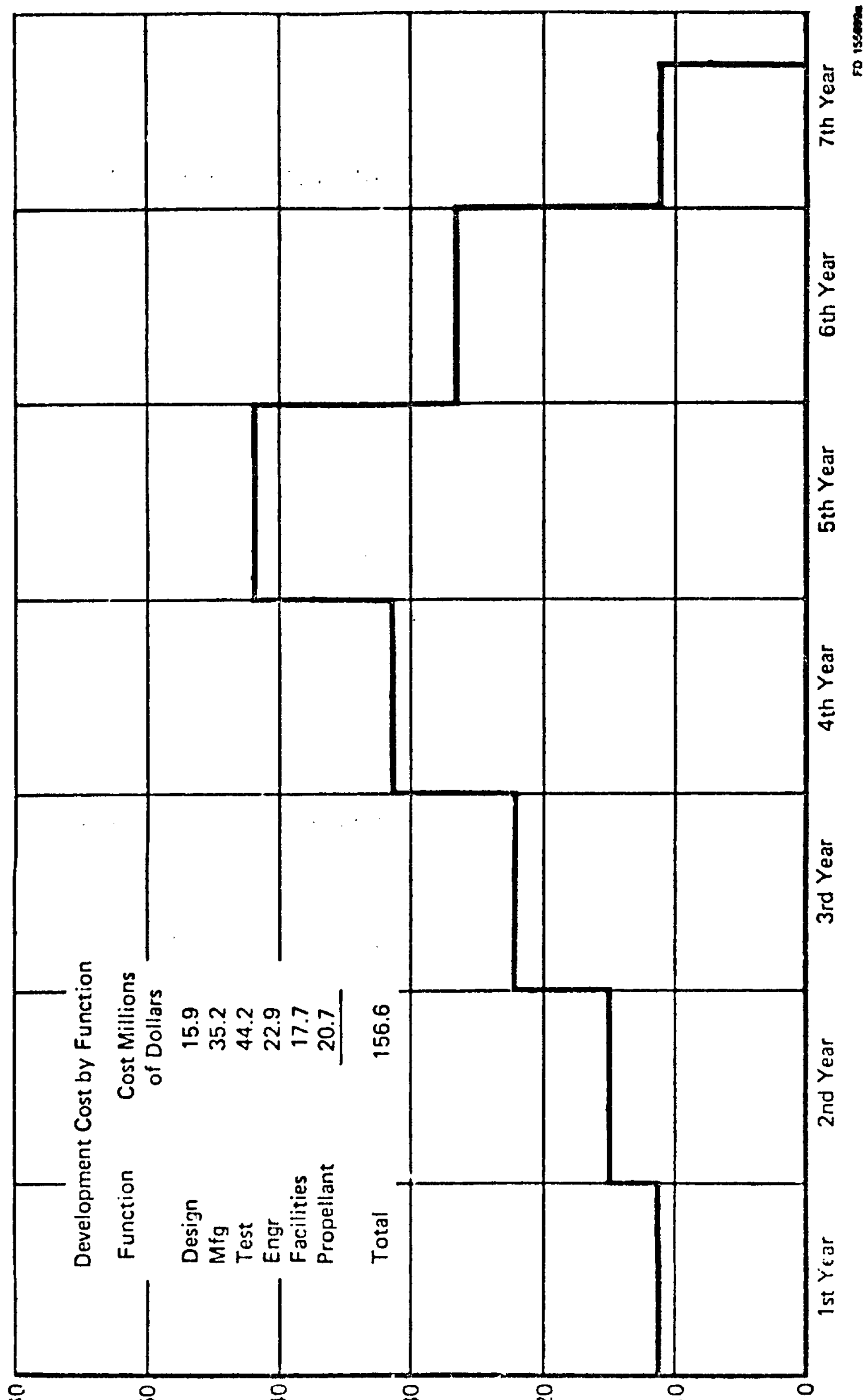
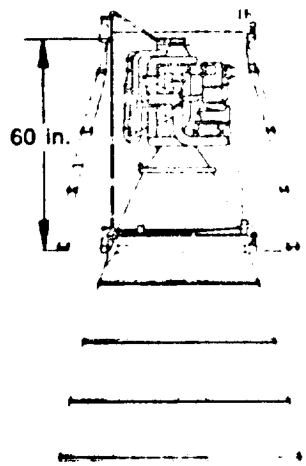


Figure 2-30. Category IV Total Development Program Cost and Funding

2.5 ADVANCED EXPANDER ENGINE



Thrust	: 15,000 lb
Chamber Pressure	: 1505 psia
Area Ratio	: 640
I_{sp}	: 482.0 sec at 6.0 MR
Operation	: Full Thrust (Low NPSH)
	: Maneuver Thrust (Saturated Propellants)
Conditioning	: Tank Head Idle
Weight	: 391 lb
Life (Design TBO)	: 300 Firings/10 hr
DDT&E Cost	: 243 Million

FD 74124b

2.5 ADVANCED EXPANDER ENGINE

2.5.1 Definitions and Requirements

Like the RL10 Category IV engine, the Advanced Expander engine is a "clean sheet" design. Unlike the Category IV engine, it is an advanced-technology engine, incorporating improved pump and turbine designs and a hydrogen regenerator. Basically, it is a "1980 state-of-the-art" design optimized specifically for use in the man-rated OTV. The baseline Advanced Expander engine has the following requirements:

1. Interface requirements: not yet defined.
2. Operating modes: Same as RL10 Derivative IIB, i.e.,
   - Tank head idle mode
   - Maneuver thrust
   - Low NPSH pumping capability at full thrust
3. Design life: 300 firings and 10 hr
4. Thrust level: 15,000 lb at 6.0 mixture ratio
5. Performance: optimize.

2.5.2 Description

The general arrangement of the Advanced Expander engine is shown in the installation drawings on Figure 2-31.

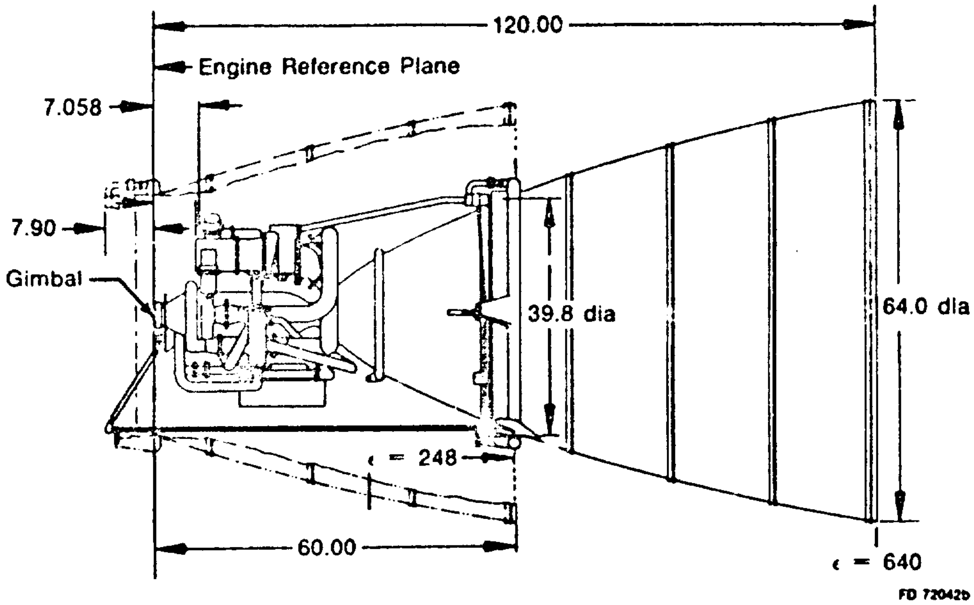


Figure 2-31. Advanced Expander Engine Installation Drawing

The principal components of the Advanced Expander cycle engine are shown in Figure 2-32. As on the current RL10, a two-stage fuel pump is used with a two-stage turbine. The fuel impellers are fully shrouded to obtain high efficiency. The single-stage oxidizer pump is driven by a single-stage turbine used in series with the fuel turbine. This was done so that the gears could be retained but the power transmitted through them could be reduced, leaving their stresses lower than in the RL10A-3-3. By gearing the pumps and inducers together, control problems during transient operation are avoided.

The combination of nickel alloy turbines and the expander power cycle give turbine life margin. To meet the thrust chamber life requirements, a milled channel chamber has been selected. The control system is also basically the same as that of the Derivative II engines, except the thrust control has been eliminated and the main fuel control valve combines the functions of the turbine bypass, and fuel vent valves. The GO_2 heat exchanger design is similar to that used on the Derivative II engines. The igniter system is of an updated lightweight design.

Chamber pressure for the advanced expander cycle is 1500 psia, which is 600 psi higher than the RL10 Category IV engine. This increase in chamber pressure is obtained by utilizing "1980 state-of-the-art" turbomachinery design (i.e., fuel pump speeds >100,000 rpm), and adding a hydrogen regenerator to increase turbine power. The hydrogen regenerator is inserted downstream of the fuel pump to increase the turbine inlet temperature by recovering heat downstream of the turbines and using it to preheat the fuel prior to cooling the thrust chamber and nozzle. The regenerator also allows a parallel chamber/nozzle coolant flow configuration to be used which, while providing adequate cooling, does not have the large pressure losses encountered in the manifolding of a counter flow configuration. The milled channel thrust chamber has been designed and the tubular primary nozzle contoured to optimize the heat transfer characteristics anticipated by the engine at the high chamber pressure. A radiation-cooled, composite material, extendible nozzle is used instead of a dump-cooled nozzle to provide a lighter, very simple system. Carbon-carbon, the composite material used, is a 1980 technology material currently used in high temperature applications because of its strength, light weight and favorable high temperature characteristics.

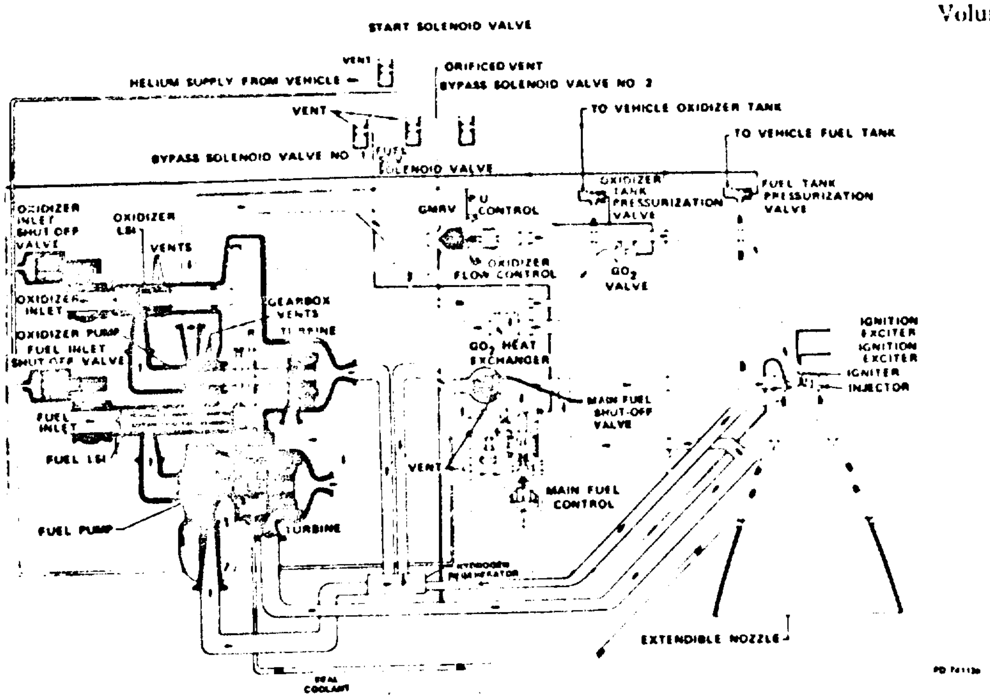


Figure 2-32. Advanced Expander Engine Propellant Flow Schematic

A simple, reliable open-loop control system is utilized for the Advanced Expander engine. The valves operate in an open-loop mode for a passive control configuration, offering an advantage in both cost and reliability over an active control configuration. The engine operates with 3% bypass margin at the design point based on RL10 production engine data that indicates this provides adequate margin for statistical deviations from nominal component operating characteristics.

The dry weight of the engine and its subassemblies are summarized in Table 2-13. Of the total engine weight of 391 lb, 74% is calculated and 26% is estimated.

TABLE 2-13. ADVANCED EXPANDER ENGINE WEIGHT

Turbopumps and Gearbox	91 lb
Thrust Chamber and Primary Nozzle	96 lb
Extendible Nozzle Actuation System	35 lb
Extendible Nozzle	16 lb
Regenerator	26 lb
GO_2 Heat Exchanger	13 lb
Controls, Valves and Actuators	68 lb
Plumbing and Miscellaneous Hardware	39 lb
Ignition System	7 lb
	391 lb

2.5.3 Operation and Performance Characteristics

2.5.3.1 Operation

The operation of this engine is basically the same as that of the Derivative IIA engine. With the start solenoid and bypass solenoid No. 1 energized, thermal conditioning in tank head idle mode is carried out with the engine operating as a pressure-fed system without turbopumps rotating. By de-energizing bypass solenoid No. 1 and energizing the other two solenoids, the main fuel shutoff valve is opened and the turbine bypass in the main fuel control valve is first closed, diverting all the fuel through the turbines, and subsequently, as turbine inlet pressure builds up, reopened to allow the engine to stabilize at maneuver thrust level. De-energizing bypass solenoid No. 2 closes this bypass to a preset area accelerating the engine to full thrust. Operation of the engine at full thrust and 6.0 mixture ratio is shown in Figure 2-33.

2.5.3.2 Engine Characteristics

The steady-state performance characteristics of the baseline Advanced Expander engine are summarized in Table 2-14.

TABLE 2-14. STEADY-STATE PERFORMANCE CHARACTERISTICS OF THE ADVANCED EXPANDER ENGINE

Operating Mode	*Tank Head Idle*	*Maneuver Thrust*	*Full Thrust*
Thrust, lb	68	1,500	15,000
Mixture Ratio	4.0	6.0	6.0
Chamber Pressure, psia	7.6	150	1505
Specific Impulse, sec	455	457.6	482.0
Fuel Turbopump Speed, rpm	0	29,550	113,825
Fuel/Oxidizer Pump Inlet Condition Limits	Superheated Phase or Liquid	0.0 NPSH	2 ft/15 ft

2.5.4 Programmatics

2.5.4.1 Engine Development

The total development program for the Advanced Expander engine requires 89 months of design, fabrication and test effort. This effort will encompass three design/build/test cycles to FFC. Figure 2-34 shows the total development program schedule and presents the major program milestones and key decision points.

Engine development is planned to be accomplished in the same manner as previously described for the Derivative II engines. Although design verification specifications (DVS's) were not formulated for the Advanced Expander engine design during this study, an estimate of verification program requirements was made by comparing the Advanced Expander engine design with the Derivative IIA engine design and deriving the design verification program for the Advanced Expander engine from that of the Derivative IIA engine.

This derived design verification program is compared with the total development program in Table 2-15.

2.5.4.2 Production and Operational Programs

Production and Operational Program costs are given in Volume III of this report.

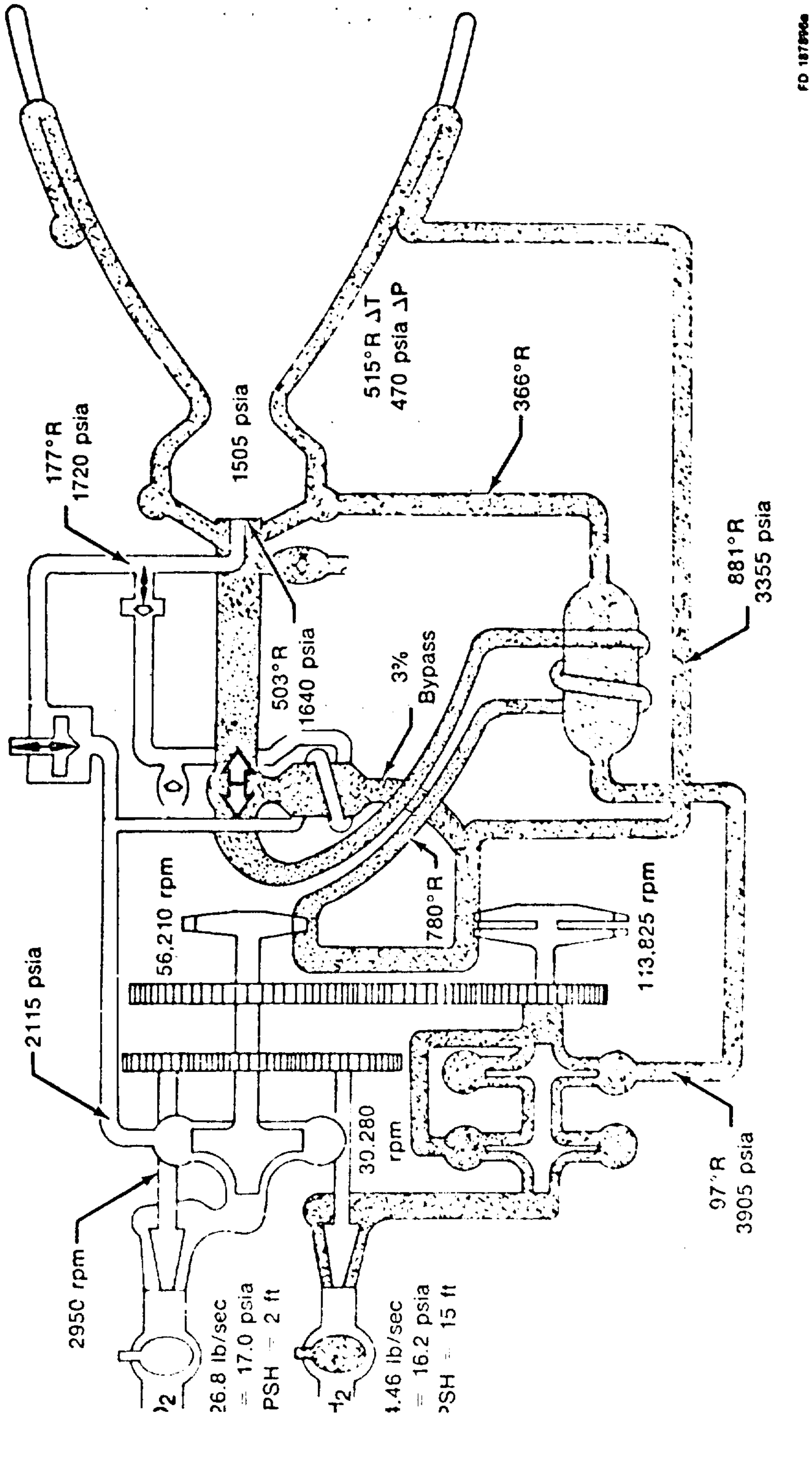


Figure 2-33. Advanced Expander Propellant Flow Schematic — Full Thrust (MR = 6.0)

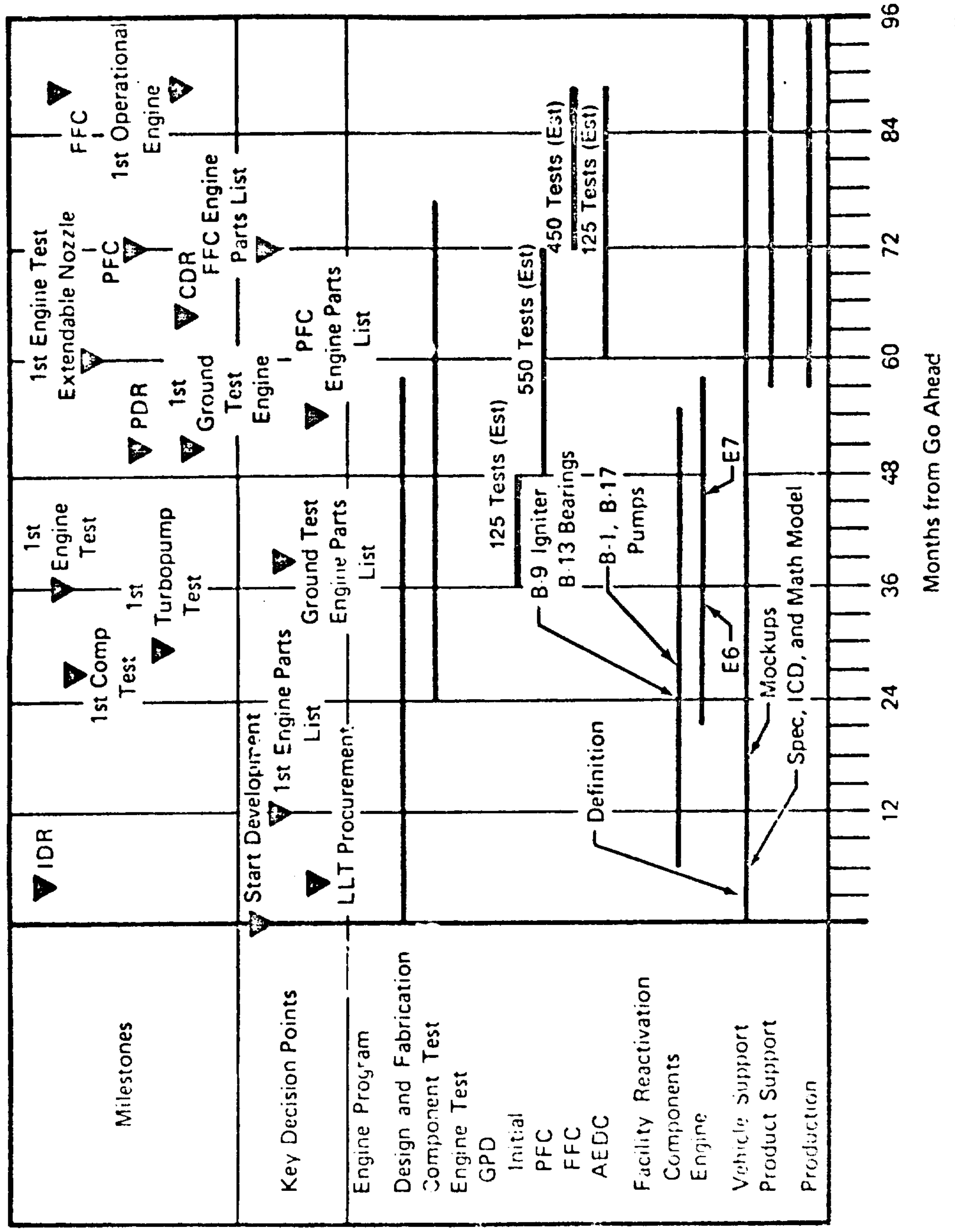


Figure 2-11. Advanced Expander Cycle Engine Development Schedule and Major Program Milestones

TABLE 2-15. COMPARISON OF DESIGN VERIFICATION PROGRAM WITH THE TOTAL DEVELOPMENT PROGRAM, ADVANCED EXPANDER ENGINE

	Total Development Program	–	*Design Verification Program*	=	*Allowance for Redesign and Verification*
Equivalent Sets of Engines	40	–	15	=	25 (63%)
Engine Builds and Rebuilds	180	–	108	=	72 (40%)
Engine Tests	1,250	–	750	=	500 (40%)
Duration to FFC in Months	89	–	77	=	12 (13%)

SECTION 3

ENGINE PARAMETRIC DATA

3.0 GENERAL

Parametric data for the engines investigated during this study, over various ranges of operating parameters, are presented in this section. The RL10 Derivative engines, defined in 1973, were reviewed and updated. The performance levels were updated to reflect improvements in JANNAF prediction techniques. These results were then adjusted to correlate with high-area-ratio nozzle performance test data generated with the RL10 and ASE engines since the study was published. Also generated were parametric engine data (performance, weight, envelope and cost) based on study ground rules (e.g., 1980 state-of-the-art, performance optimized, man-rated reliability) for advanced-expander and staged-combustion cycle engines. Preliminary cycle studies were conducted which defined ground rules, and a viable engine configuration was selected for each basic cycle. The parametric data was then generated using these basic configurations as starting points. In addition, an analysis was conducted to determine the effect of variations in pump inlet net positive suction head (NPSH) on engine weight and inlet line interface.

3.1 RL10 DERIVATIVE ENGINE PARAMETRIC DATA

Four baseline engines were defined under this and previous contracts. One of them is a new design (RL10 Category IV) and the other three are derivatives of the existing RL10 engine (RL10 Derivatives IIA, IIB and IIC). All of the baseline engines have been designed according to fixed requirements. Since these requirements are not optimum for all vehicle applications, a parametric analysis was conducted. Table 3-1 summarizes the independent and dependent parameters which were perturbated in this analysis.

TABLE 3-1. PARAMETRIC STUDY VARIABLES

Baseline Engine	*Independent Variable*	*Dependent Variable*	*Constant*
RL10 Derivatives IIA, IIB and IIC	Installed Length (70 to 55 in.)	Specific Impulse Engine Weight Engine Geometry	Thrust (15,000 lb) Mixture Ratio (6.0) Chamber Pressure (400 psia)
	Mixture Ratio (4.5 to 7)	Specific Impulse Engine Weight Engine Geometry	Thrust (15,000 lb) Installed Length (55 and 70 in.) Chamber Pressure (400 psia)
RL10 Category IV	Installed Length With RL10 Derivative IIA Interfaces (50 to 70 in.)	Specific Impluse Engine Weight Engine Geometry Chamber Pressure	Thrust (15,000 and 20,000 lb) Mixture Ratio (6.0) Life (10 hr/300 firings)
	Mixture Ratio (4.5 to 7)	Specific Impulse Engine Weight Engine Geometry Chamber Pressure	Thrust (15,000 lb) Installed Length (57 in.) Life (10 hr/300 firings)

Each parametric engine design was optimized to give maximum nozzle performance for a given length (minimum length contour). Two-dimensional equilibrium analyses were used to parametrically define minimum length nozzle contours. Performance was calculated using JANNAF methodology and adjusted for demonstrated high expansion ratio nozzle performance.

The nozzles were of the two-position variety in which the primary (fixed) portion of the nozzle was regeneratively cooled and the secondary (translating) portion of the nozzle was radiation cooled. The propellant injector was considered to be optimized for each design point condition for the Derivatives IIA, IIB and Category IV engines, and was assumed to give an energy release efficiency of 0.9970 and a striation loss of less than 0.1 sec of impulse. The propellant injector was not changed from the current RL10A-3-3 design for the Derivative IIC engine.

Engine weight was generated using the turbopump, nozzle actuator, heat exchanger, control, plumbing and miscellaneous weights from the 15,000 lb thrust baseline engine design. The remaining engine weight, i.e., chamber/nozzle weight, was estimated using a fixed weight/ft^2 of nozzle surface area of 0.37, 2.46 and 6.0 for radiation-cooled, regeneratively cooled tubular and regeneratively cooled nontubular sections, respectively.

3.2 DERIVATIVE II ENGINE PARAMETRIC DATA

The following ground rules were used in the parametric study of Derivative II engines:

1. Constant design thrust of 15,000 lb
2. Constant chamber pressure of 400 psia for design point analysis
3. Engine mixture ratio $4.5 \leq O/F \leq 7.0$
4. Retracted engine length varied between 55 and 70 in.
5. Primary nozzle exit diameter = 40 in.

3.2.1 Retracted Engine Length Effects

The available range of retracted engine lengths for the Derivative II engine is limited on the low end by engine powerhead diameter and on the high end by the 70-in. length constraint inherent in the engine definition. The minimum retracted engine length is essentially made up of gimbal length, combustion chamber length, and primary nozzle length. The turbopumps are packaged around the combustion chamber and do not add to the overall length.

Since the gimbal and combustion chamber lengths remain essentially constant, engine length is primarily a function of primary nozzle area ratio; and since engine thrust, chamber pressure and nozzle contour are almost constant, minimum engine length is primarily a function of primary nozzle exit diameter. The smaller the exit diameter, the shorter the engine length. The primary nozzle exit diameter must be slightly larger than the engine powerhead diameter, however, to allow the secondary nozzle to fully translate.

The shortest allowable Derivative II engine has a retracted length of 55 in. The engine was configured with a secondary nozzle length equal to the length of the primary engine. An "equal length" engine shorter than this would require the primary nozzle diameter to be less than 40 in. violating ground rule "5" given above. The 40-in. minimum diameter constraint is determined by the Derivative II engine powerhead diameter of 39.5 in.

For engines longer than 55 in. the best performance is obtained by "equal length" configurations which require the primary nozzle exit diameter to be greater than 40 in. For example, a 70-in. engine, with total engine length (secondary nozzle extended) of 140 in., would have a primary exit diameter of approximately 57.5 in. Testing of such an engine would require a modification to the GPD test facilities, however, since the E-6 stand will currently accept nozzle diameters only slightly greater than 40 in. and the E-7 stand will accept nozzle diameters up to approximately 47.5 in. Figure 3-1 presents design parametric data at a mixture ratio of 6.0 for retracted length variations between 55 and 70 in. with the primary nozzle exit diameter held at 40 in. throughout to allow testing on GPD stands.

3.2.2 Engine Mixture Ratio Effects

Design parametric performance data were generated over the range of mixture ratios for engines designed at the minimum and maximum allowable retracted engine lengths. The engine designs presented are capable of providing an additional off-design mixture ratio excursion of plus or minus one-half mixture ratio unit. Figures 3-2 and 3-3 present parametric design point data over mixture ratio range of 4.5 to 7.0, respectively, for 55 and 70 in. retracted length engines.

Off-design specific impulse vs mixture ratio characteristics for 55 in. retracted length Derivative II engines (Figure 3-4) are provided for reference purposes.

3.3 CATEGORY IV

The ground rules used in the generation of the Category IV parametric data are as follows:

1. Optimum chamber pressure for each design point
2. Optimum chamber length for each design point; however, chamber length should not be shorter than 12 in. to ensure good combustion performance, nor greater than 30 in.
3. Evaluate thrust levels of 15,000 and 20,000 lb
4. Engine mixture ratio $4.5 \leq O/F \leq 7.0$
5. Retracted engine length between 70 in. (RL10A-3-3 length) and the minimum set by the powerhead diameter
6. Engines interchangeable with RL10 Derivative IIA; powerhead diameter limits the primary nozzle exit diameter to a level greater than 37 in.

Regenerative nozzle parameters which affect engine power level (ΔP and ΔT) were estimates relative to the baseline design point and were used to determine the optimum chamber pressure and chamber length for each design evaluated. The optimization technique was the same as that used to optimize the Category IV baseline design as described in Section 3.3, Volume II of Final Report FR-6011, "Design Study of RL10 Derivatives."

3.3.1 Engine Length Effects

The shortest permissible engine design had a retracted length of 50.5 in. and 50.0 in. for respective thrust levels of 15,000 and 20,000 lb. These engines had chamber lengths of 12 in. Any engine design shorter than this would require either a chamber length shorter than 12 in. or a primary nozzle exit diameter less than 37 in., violating either ground rule "2" or "6." Figure 3-5 presents design parametric data at a mixture ratio of 6.0 for retracted length variations between 50 and 70 in. for thrust levels of 15,000 and 20,000 lb.

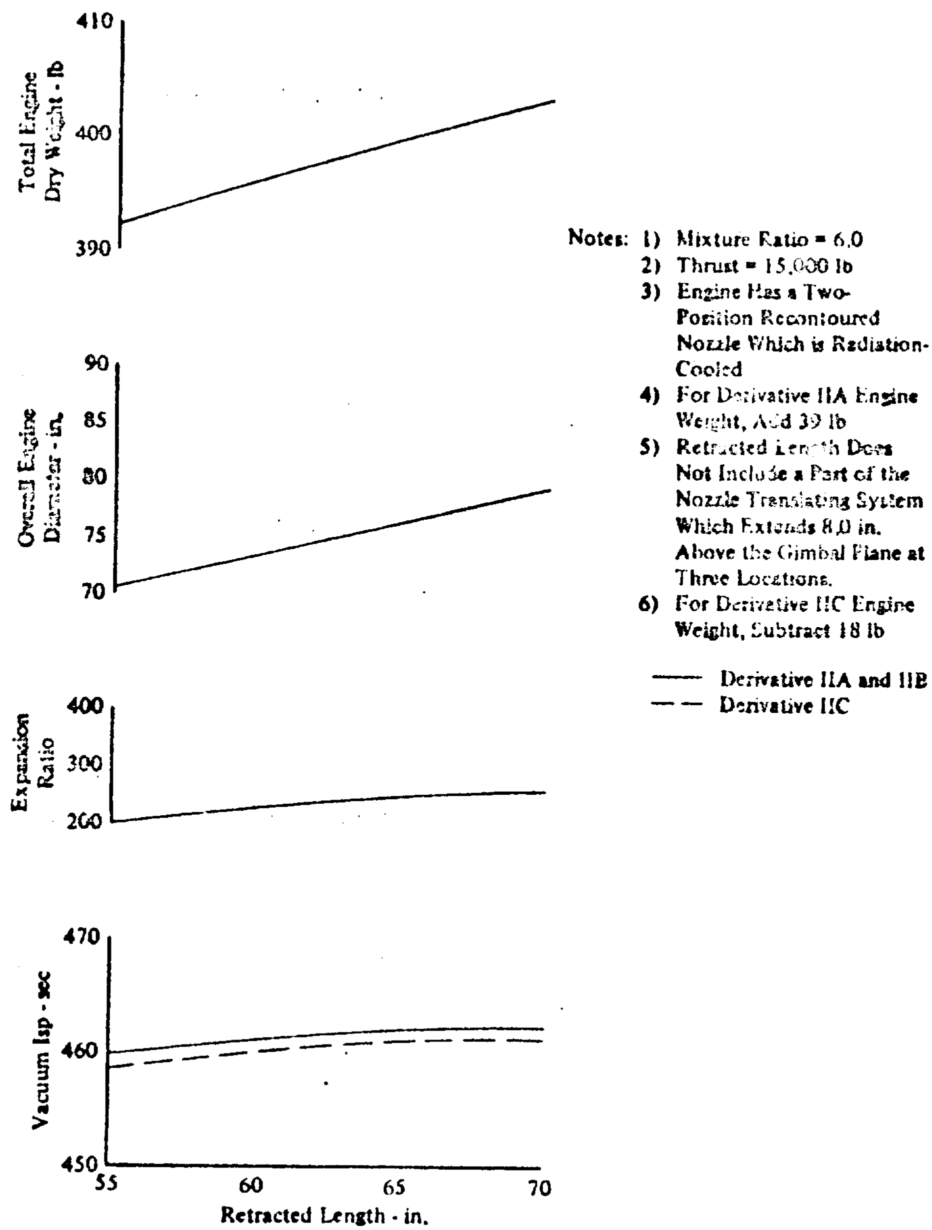


Figure 3-1. Effect of Retracted Length on RL10 Derivative II Engine Design Point Performance (Primary Nozzle Diameter Limited to 40 in.)

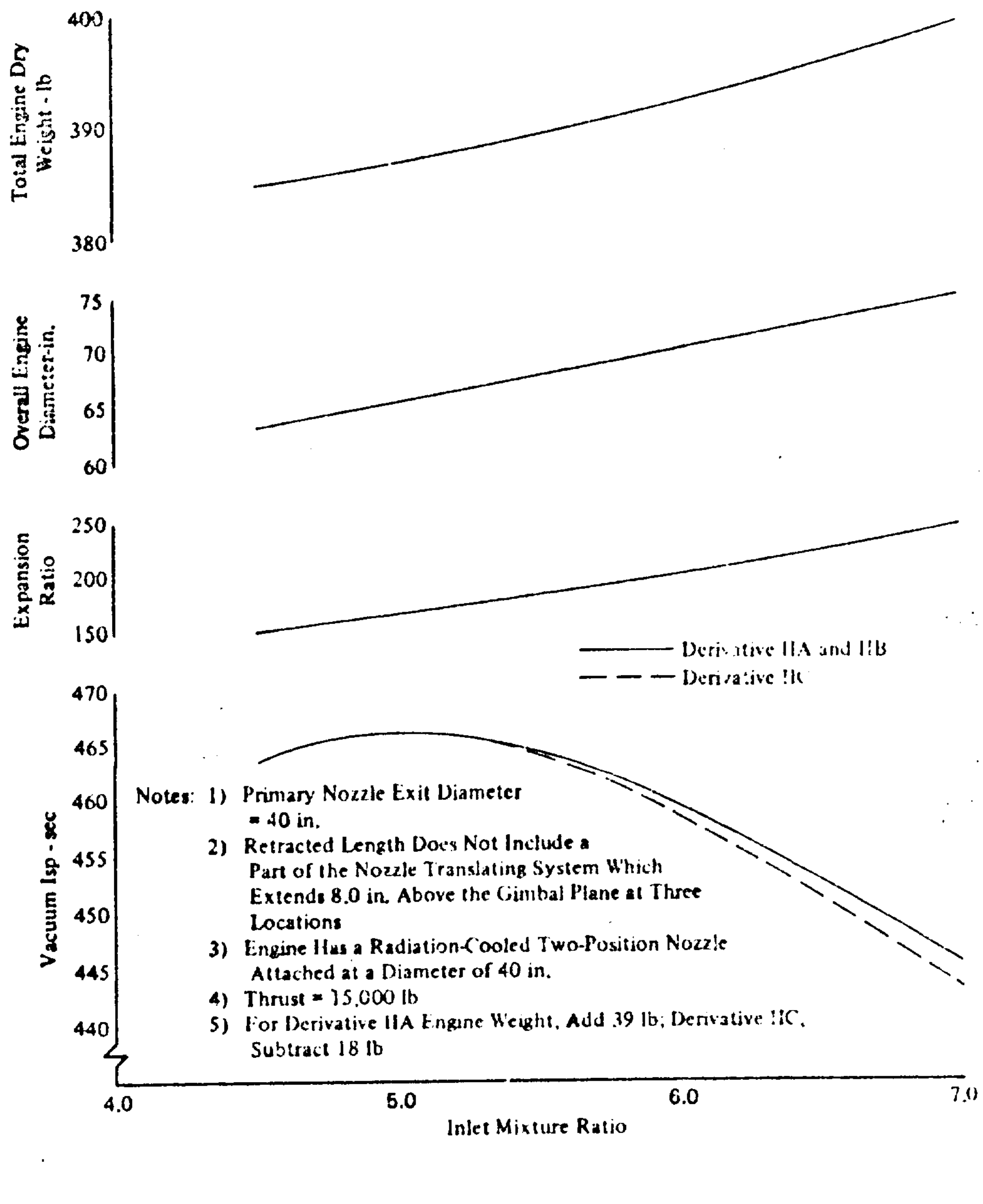


Figure 3-2. Effect of Mixture Ratio on RL10 Derivative II Engine Design Point Performance (Retracted Length = 55 in.)

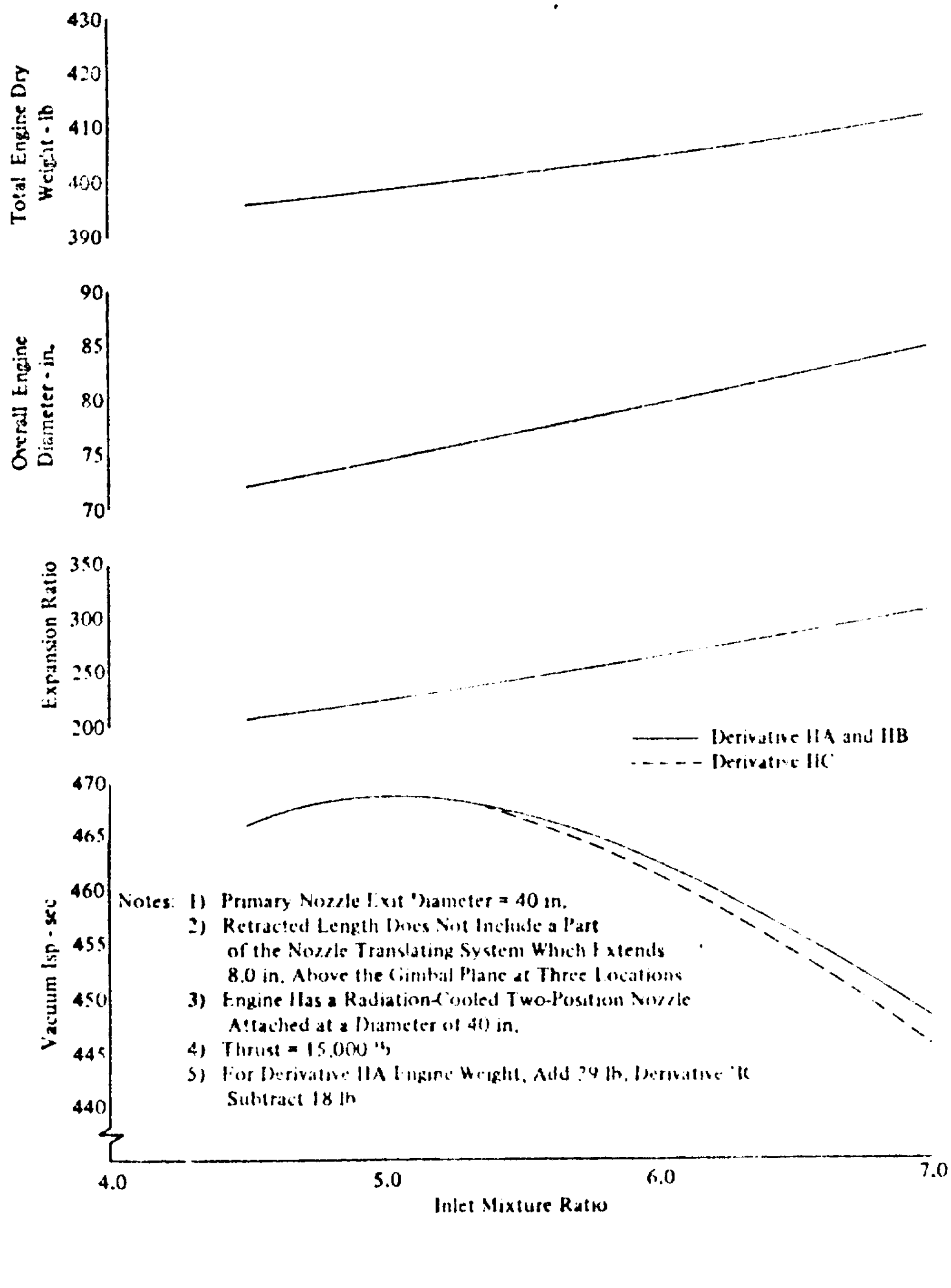


Figure 3-3. Effect of Mixture Ratio on RL10 Derivative II Engine Design Point Performance (Retracted Length = 70 in.)

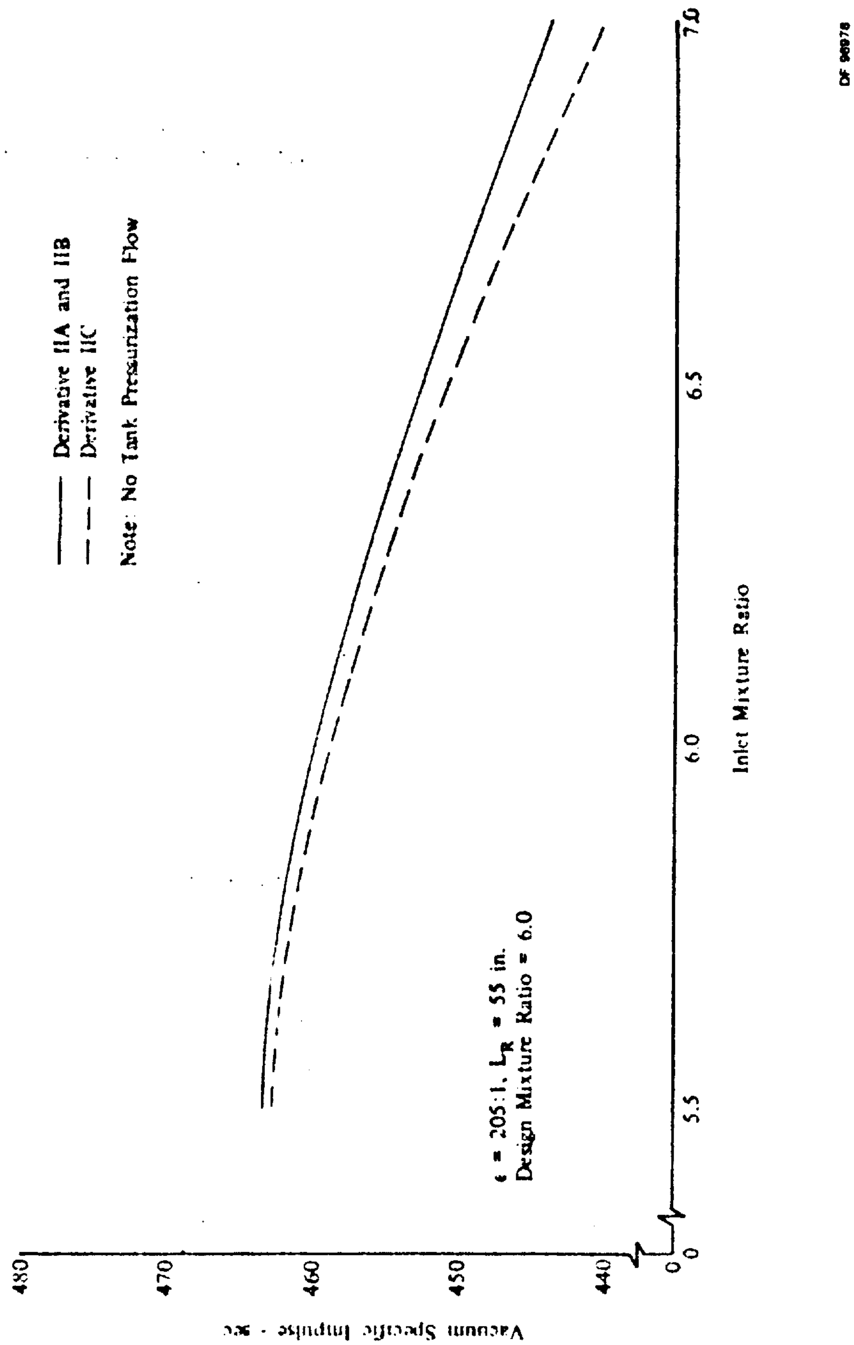


Figure 3-4. Estimated Effect of Inlet Mixture Ratio on Vacuum Specific Impulse, Derivative II Engine — Full Thrust

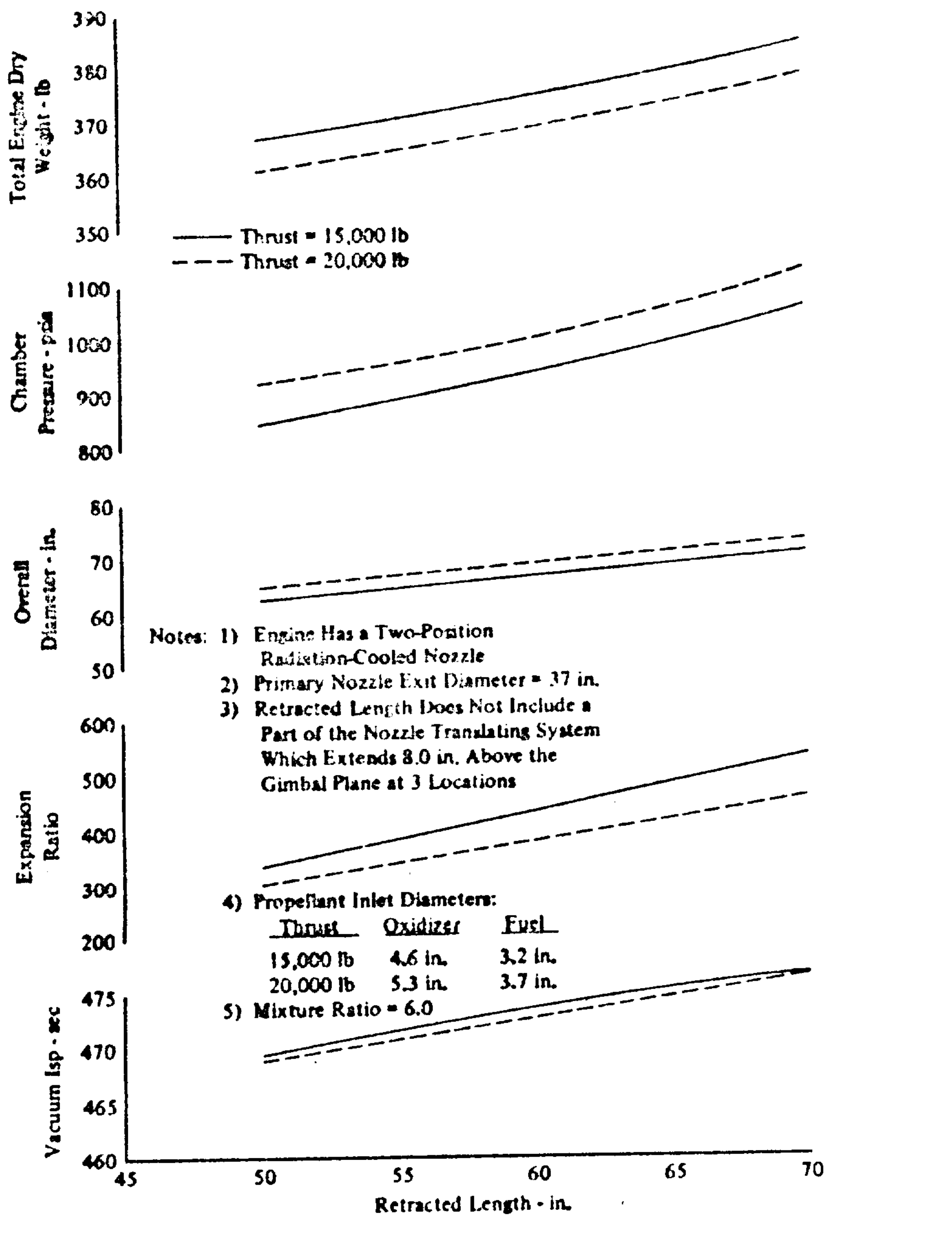


Figure 3-5. Effect of Retracted Length on RL10 Category IV Engine Design Point Performance

3.3.2 Mixture Ratio Effects

Figure 3-6 represents parametric performance data over an engine mixture range of 4.5 to 7.0 for a 15,000 lb thrust Category IV engine having a retracted length of 57 in. Figure 3-7 gives off-design specific impulse characteristics for a 57 in. retracted length engine which has a design point mixture ratio of 6.0.

3.4 ADVANCED ENGINE PARAMETRIC DATA

Parametric Engine Data (performance, weight, envelope and cost) were generated at 10K, 20K and 30K lb thrust levels for retracted engine lengths of 55, 60, and 65 in. for advanced expander and staged-combustion cycle engines. These engine designs were consistent with the OTV engine requirements (as expressed in the contract Statement of Work) listed below:

OTV Engine Requirements

1. The engine will operate on liquid hydrogen and liquid oxygen propellants.

2. Engine design and materials technology are to be based on 1980 State of the Art.

3. The engine must be capable of accommodating programmed and/or commanded variations in mixture ratio over an operating range of 6:1 to 7:1 during a given mission. The effects on engine operation and lifetime must be predictable over the operating mixture ratio range.

4. The propellant inlet temperatures shall be 162.7°R for the oxygen boost pump and 37.8°R for the hydrogen boost pump. The boost pump inlet NPSH at full thrust shall be 2 ft for the oxygen pump and 15 ft for the hydrogen pump.

5. The service-free life of the engine cannot be less than 60 start/shutdown cycles or 2 hr accumulated run time, and the service life between overhauls cannot be less than 300 start/shutdown cycles or 10 hr accumulated run time. The engine shall have provisions for each of access, minimum maintenance, and economical overhaul.

6. The engine, when operating within the nominal prescribed range of thrust, mixture ratio, and propellant inlet conditions, shall not incur during its service life, chamber pressure oscillation, disturbances, or random spikes greater than ±5% of the mean steady-state chamber pressure.

7. The engine nozzle is to be a contoured bell with an extendible/retractable section.

8. Engine gimbal requirements are +15 deg and −6 deg in the pitch plane and ±6 deg in the yaw plane.

9. The engine is to provide gaseous hydrogen and oxygen autogenous pressurization for the propellant tanks.

10. The engine is to be man-rated.

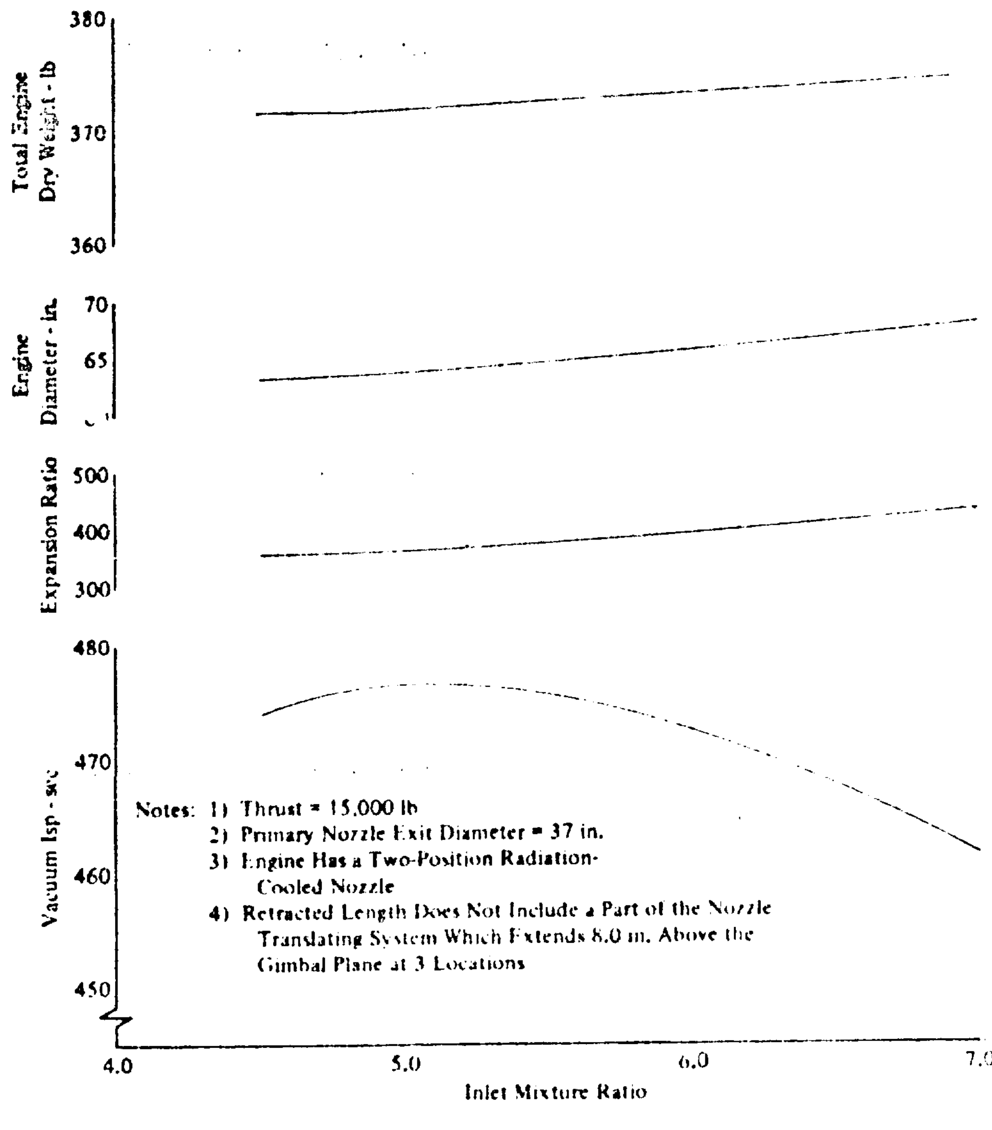


Figure 3-6. Effect of Mixture Ratio on RL10 Category IV Engine Design Point Performance (Retracted Length = 57 in.)

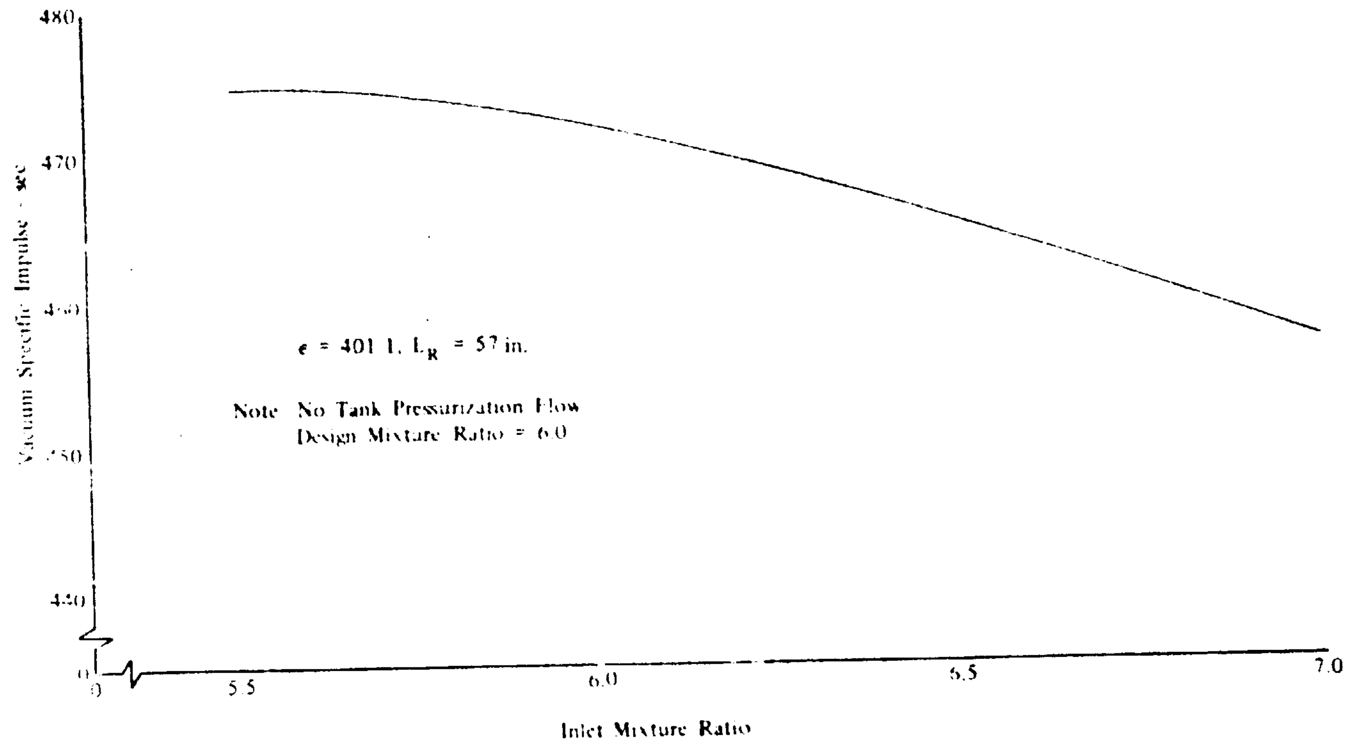


Figure 3-7. Estimated Effect of Inlet Mixture Ratio on Vacuum Specific Impulse, Category IV Engine — Full Thrust

3.4.1 Design Point Data

Each parametric engine design was optimized to give maximum nozzle performance for a given length. Performance predictions were made using JANNAF methodology adjusted for RL10 and ASE high area ratio test data.

The nozzles are all two-position configurations with the primary nozzle regeneratively cooled and the secondary nozzle (extendable section) radiation cooled. The propellant injector was considered to be optimized for each design point condition, and was assumed to give an energy release efficiency of 0.997 and a striation loss of less than 0.1 sec of impulse.

Ground rules for both engine cycles are presented in Table 3-2. Conservative levels relative to 1980 State of the Art were used used for most parameters to provide parametric results that would be representative of those that could be realistically achieved in an engine development program.

TABLE 3-2. PARAMETRIC STUDY GROUND RULES FOR OTV ENGINE DESIGN POINTS

	Advanced Expander Engine	*Staged Combustion Engine*
Fuel Pump Suction Specific Speed	13,000	13,000
Fuel Pump Speed — rpm	115,000	115,000
Fuel Pump Tip Speed (Maximum) ft/sec	2,200	2,200
Oxidizer Pump Suction Specific Speed	20,000	20,000
Oxidizer Pump Specific Speed	1,420	1,420
Turbine Bypass Flow — %	3	0
Turbine Configuration	Series	Parallel
Turbine Inlet Temperature — °R		1,860
Minimum Turbine Blade Height — in.	0.3	0.3
Minimum Turbine Vane Angle — deg	15	15
Fuel Leakage Flow — lb/sec	0.1	0.1
Fuel Inlet NPSH — ft	15	15
Oxidizer Inlet NPSH — ft	2	[illegible]
H₂ Regenerator Effectiveness	[illegible]	
Chamber Length — in.	[illegible]	[illegible]

Chamber pressure characteristics for the two engine cycles are shown in Figures 3-8 and 3-9 as a function of retracted engine length and thrust level. Chamber pressure increases as retracted length is increased for the advanced expander cycle because of the additional turbine power obtained from the increased primary nozzle surface area.

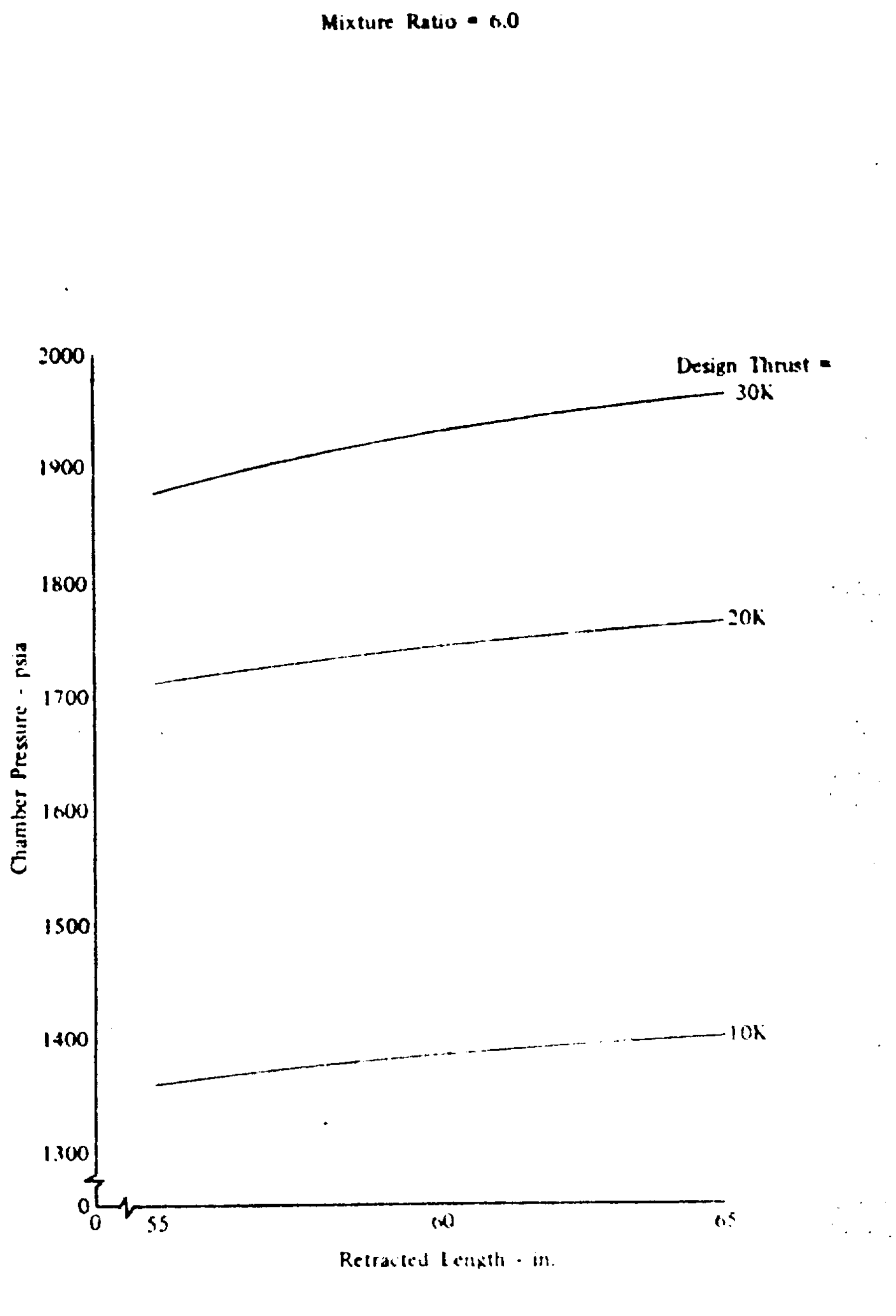


Figure 3-8. Advanced Expander Cycle Chamber Pressure Characteristics

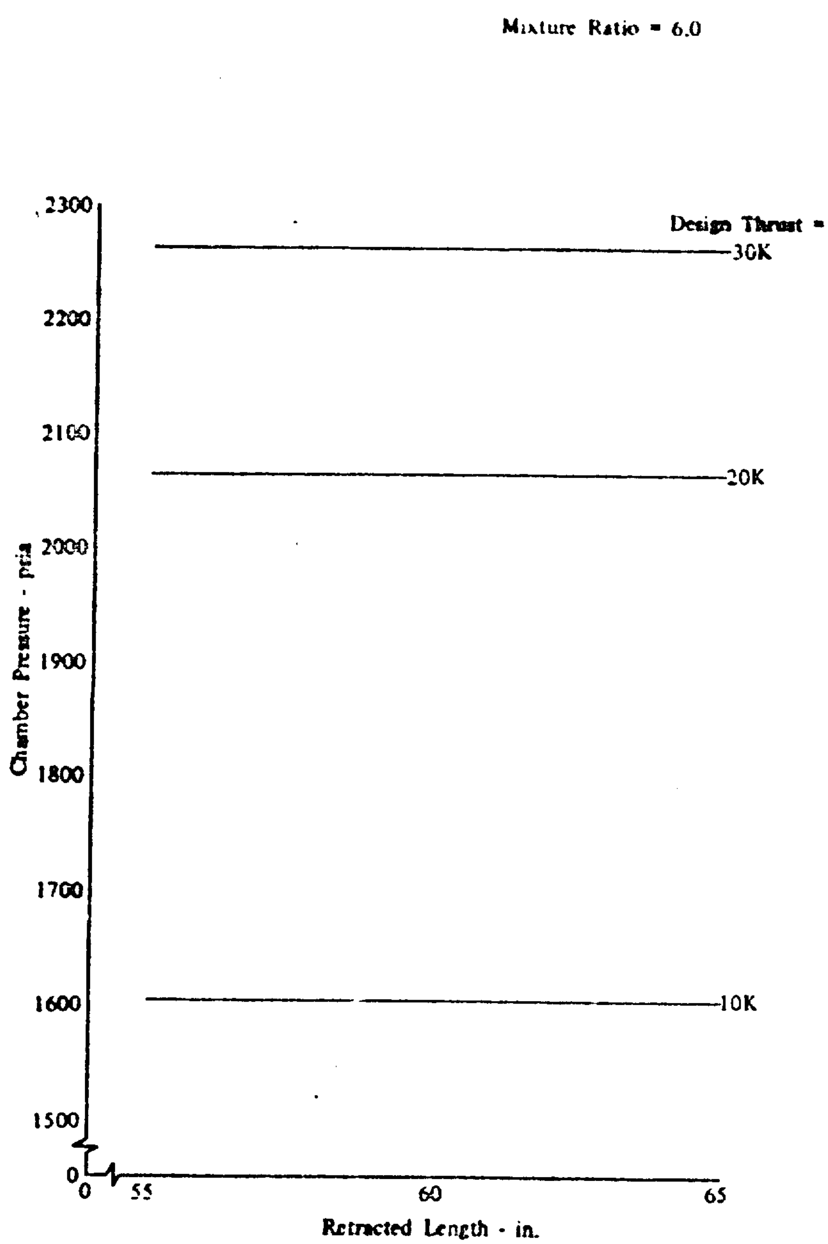


Figure 5-9. Staged Combustion Cycle Chamber Pressure Characteristics

Staged-combustion cycle chamber pressure is constant as a function of retracted length, because the increased heat transfer and pressure loss has no significant effect on available turbine power. Higher chamber pressure results from increasing thrust for both cycles, because of the improved component efficiencies attainable with the higher flowrates. Turbine power (and therefore chamber pressure) increases for the expander cycle as thrust is increased, since the effect of the improved component efficiencies is greater than the effect of the reduced turbine inlet temperature. For the staged combustion cycle, the chamber pressure levels are limited by the pump tip speed limit (which limits pump discharge pressure for a given speed) rather than component efficiencies.

A payload capability optimization study for the advanced expander cycle engine showed maximum performance at the maximum extended length for the three specified engine retracted lengths as indicated in Figure 3-10. A similar study for the staged-combustion engine showed that, because of its higher chamber pressure levels and shorter combustion chamber, area ratios are already so high that increases in engine length only provide a slight increase in specific impulse, which is more than compensated for by the increase in engine weights. Therefore, maximum performance was achieved at less than the maximum extended length for the 60 and 65 in. retracted lengths as shown in Figure 3-11.

Figures 3-12 through 3-14 present the advanced expander cycle design point (O/F = 6:1) parametric performance as a function of engine retracted length and thrust level. Figures 3-15 through 3-17 present the same information for the staged-combustion cycle.

3.4.2 Minimum NPSH Effects

The effect of minimum net positive suction head (NPSH) on engine inlet diameter and engine weight was defined for thrust levels of 10K, 20K, and 30K at a mixture ratio of 6.0:1. Figure 3-18 presents the effect of oxidizer NPSH variation over a range of 0 to 16 ft on oxidizer low-pressure pump inlet diameter and engine weight. Figure 3-19 presents equivalent information for the fuel side for an NPSH range of 0 to 60 ft. Fuel NPSH has a minor effect on both inlet line diameter and engine weight. However, the oxidizer NPSH effect becomes significant as NPSH approaches zero. Therefore, it may be more payload effective to provide oxidizer tank pressurization capability than to design the engine low-pressure oxidizer pump for NPSH levels of <2 ft.

3.4.3 Off-Design Parametric Data

Off-design parametric data has been generated for both the advanced expander and the staged-combustion engine cycles. Off-design performance and critical engine cycle parameters were defined for an inlet mixture ratio excursion of 6.0 to 7.0 for optimized 60-in. retracted length engines at thrust levels of 10K, 20K, and 30K lb. Advanced expander engine off-design characteristics are shown in Figures 3-20 to 3-22, while Figures 3-23 to 3-25 depict the staged-combustion engine off design characteristics.

During off-design operation, chamber pressure is held constant with mixture ratio. Engine off design performance characteristics are defined using JANNAF methodology and correlations based on RL10 and ASE test data. Off-design engine cycle parameter characteristics for the optimized 55- and 65-in. retracted length engines are not defined, because the internal engine parameters do not change appreciably. However, engine performance is affected by engine length, and so, off-design performance is defined for all three retracted lengths for the two engine cycles and is shown in Figures 3-26 and 3-27.

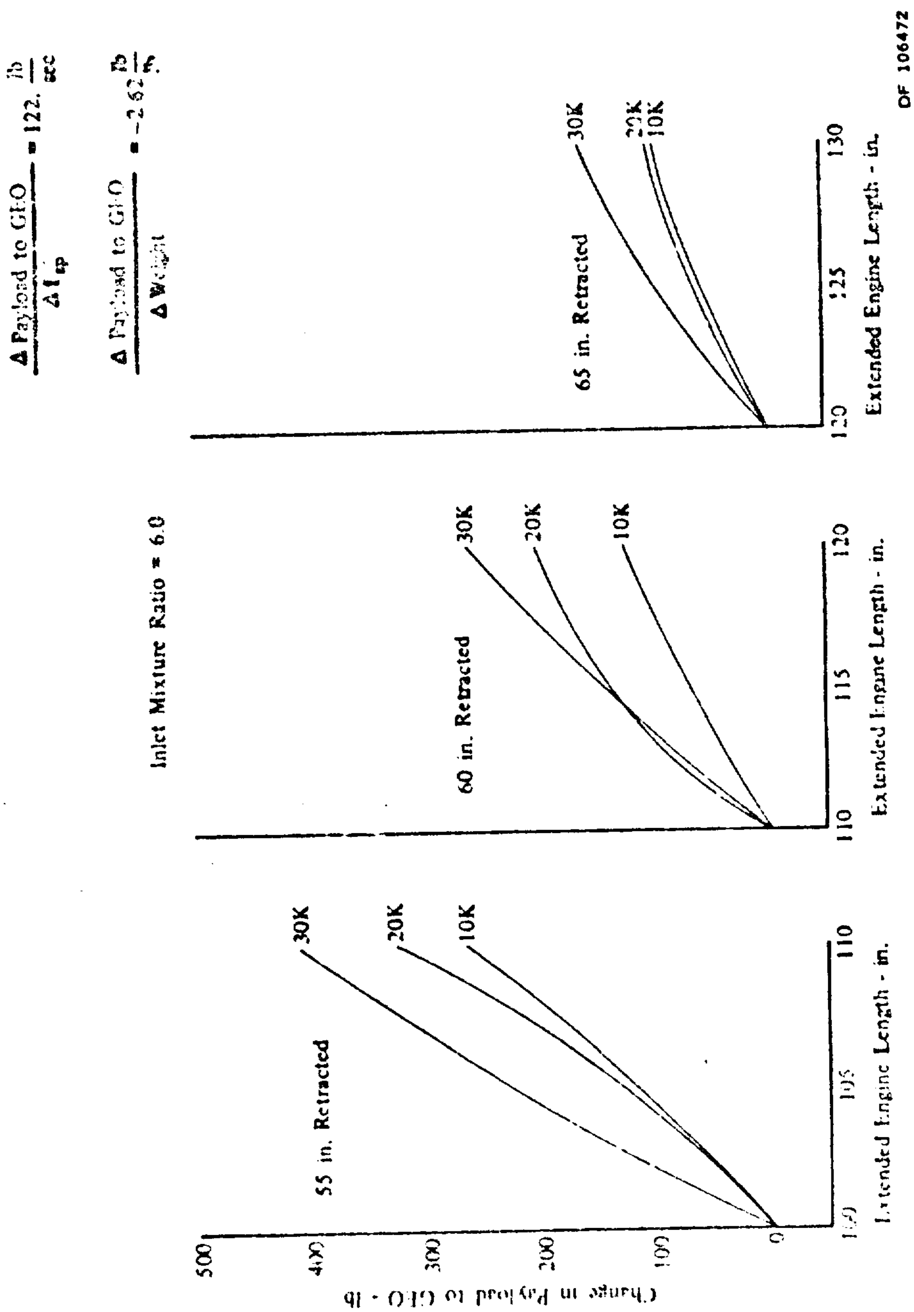


Figure 3-10. Advanced Expander Engine Length Optimization

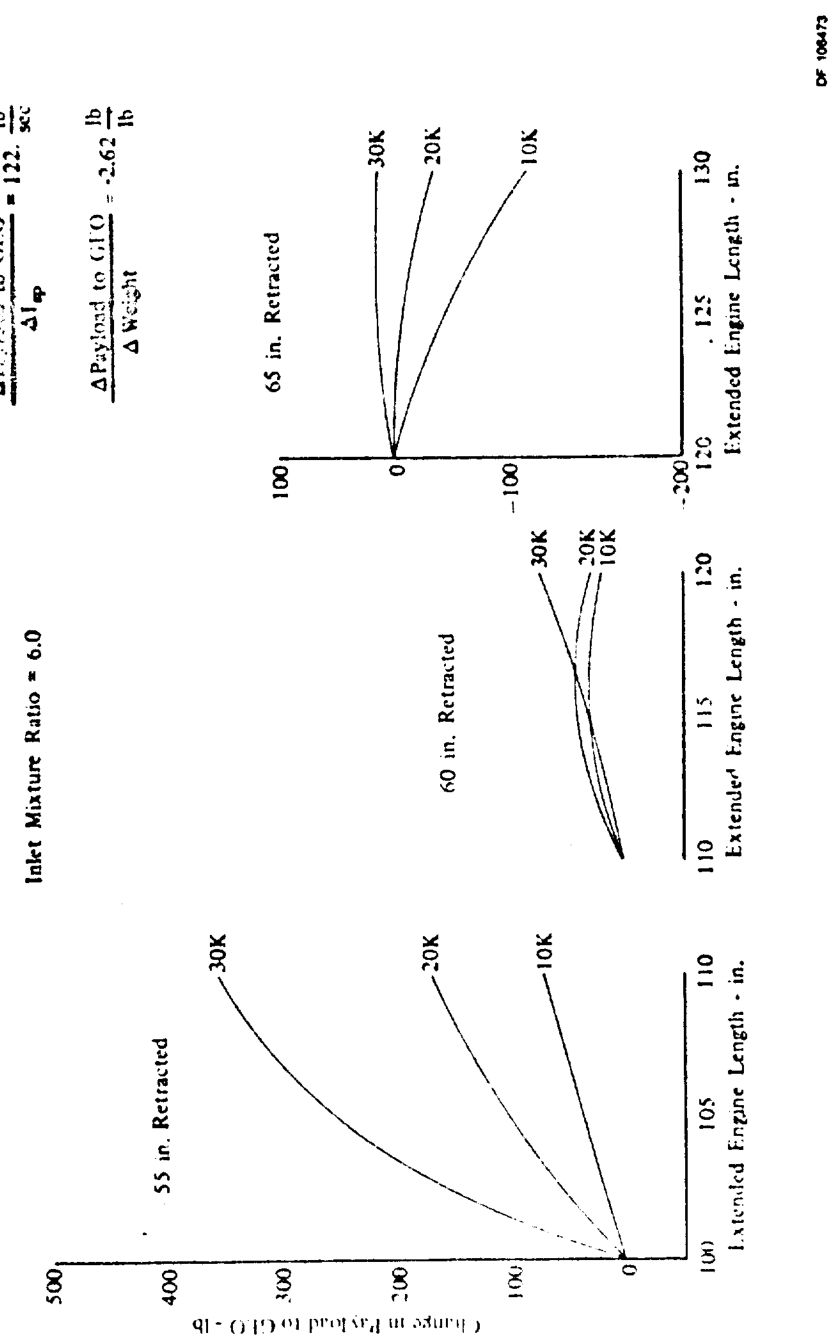


Figure 3-11. Staged Combustion Engine Length Optimization

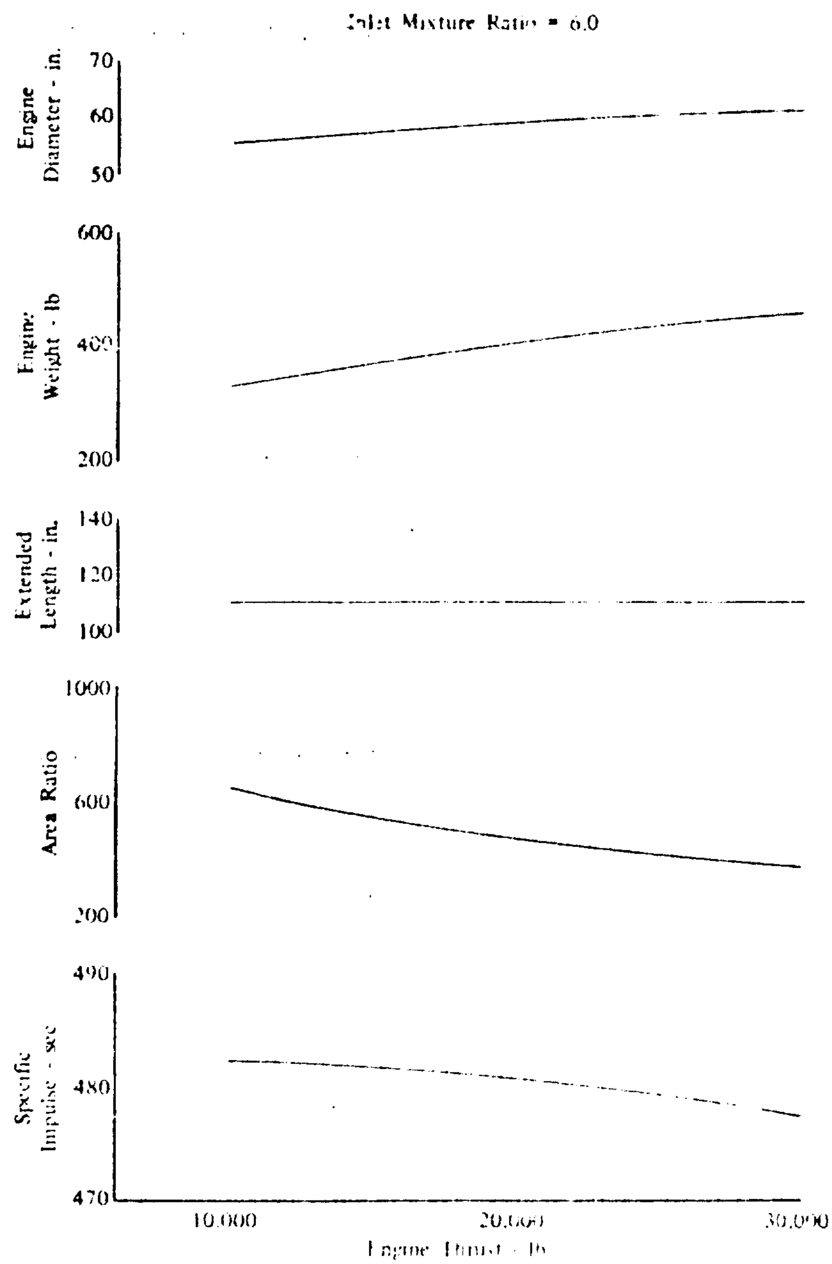


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Figure 3-12. Advanced Expander Engine Parametric Characteristics (55 in Retracted Length)

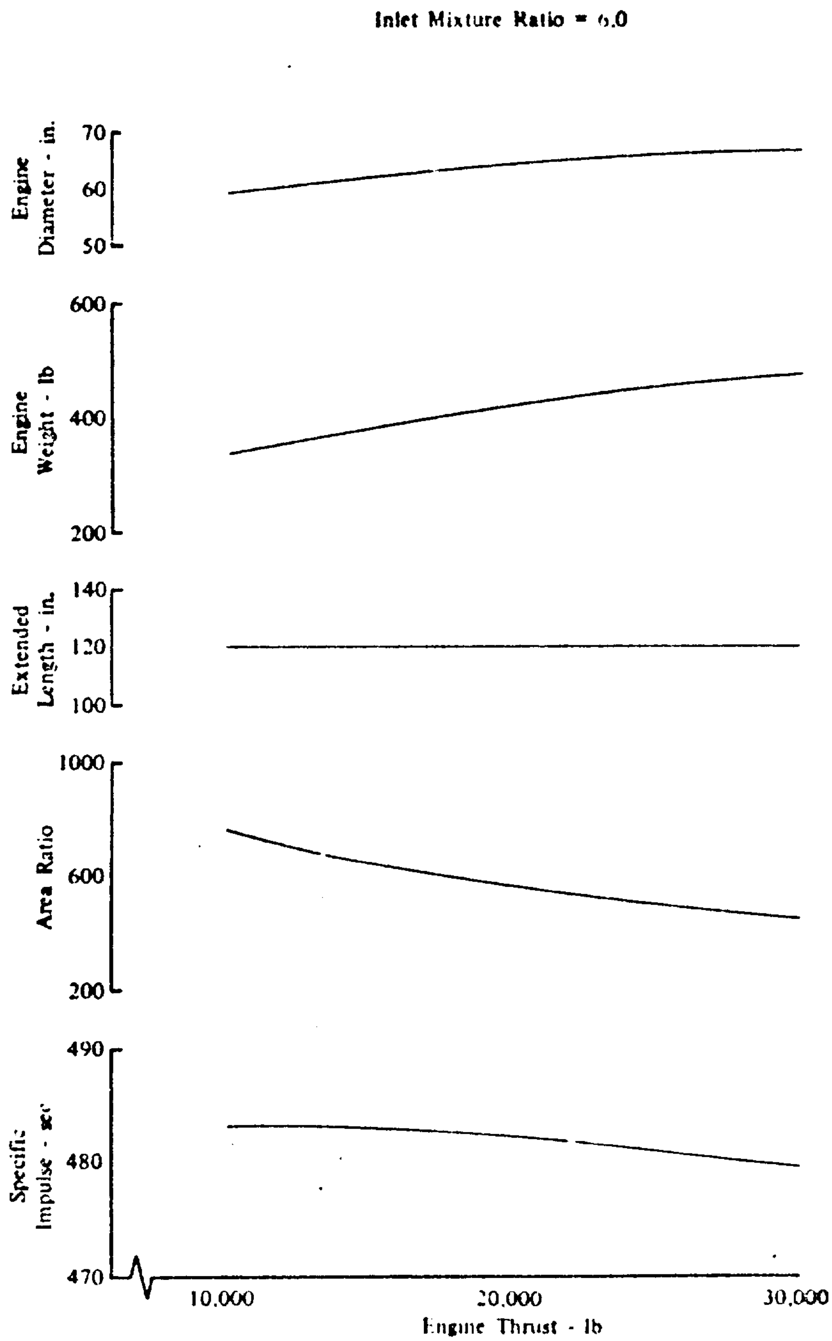

Figure 3-13. Advanced Expander Engine Parametric Characteristics (60 in. Retracted Length)

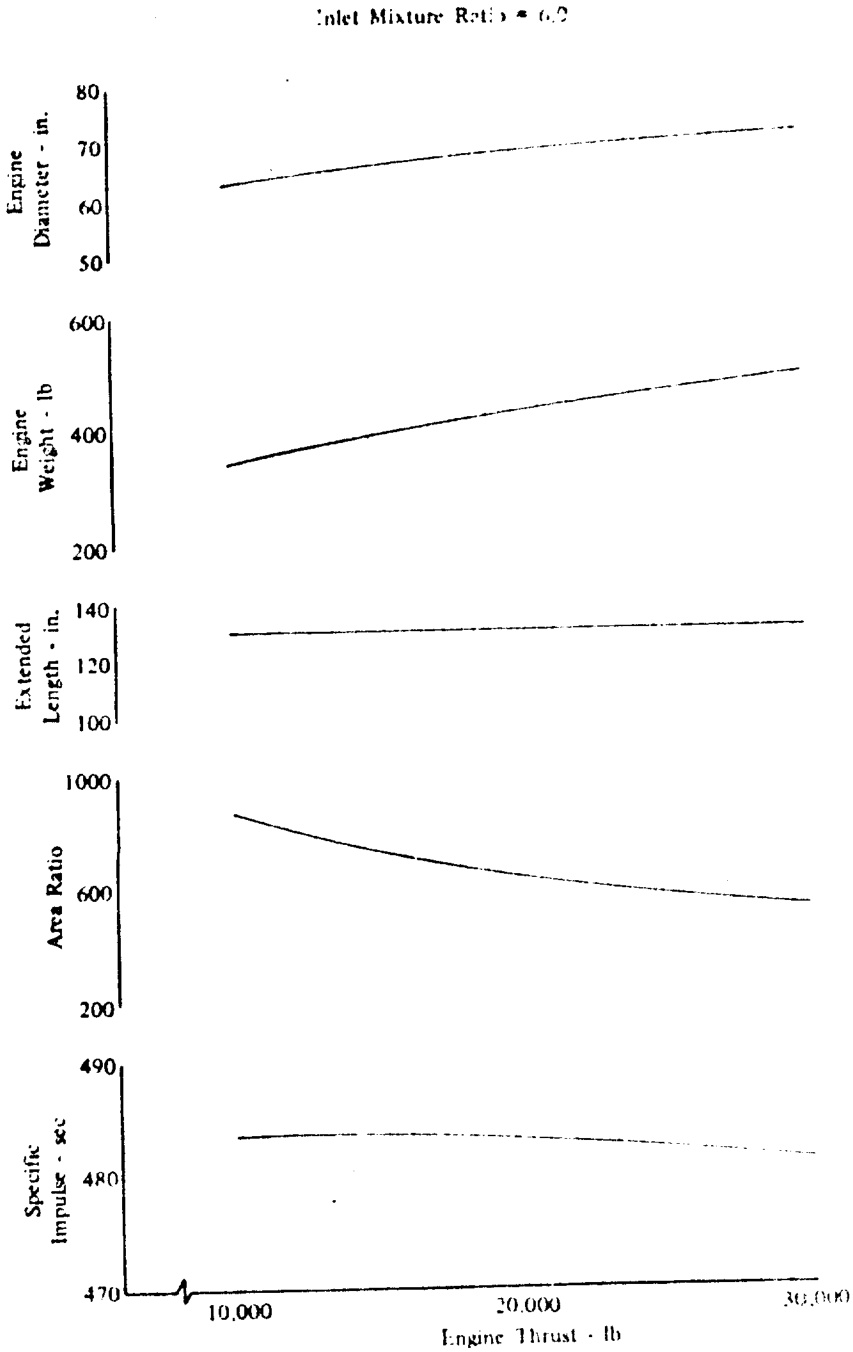

Figure 3-11. Advanced Expander Engine Parametric Characteristics (65 in Retracted Length)

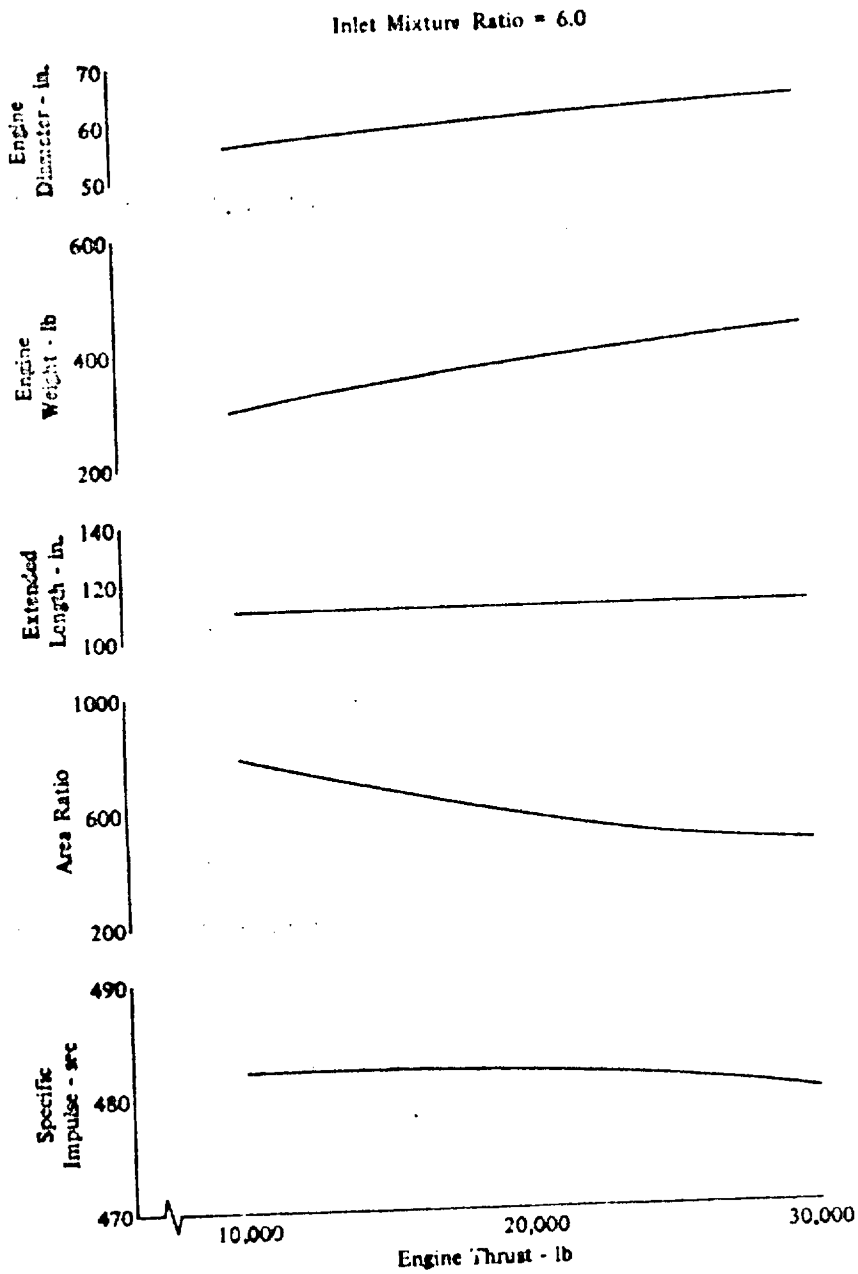

Figure 3-15. Staged Combustion Engine Parametric Characteristics (55 in. Retracted Length)

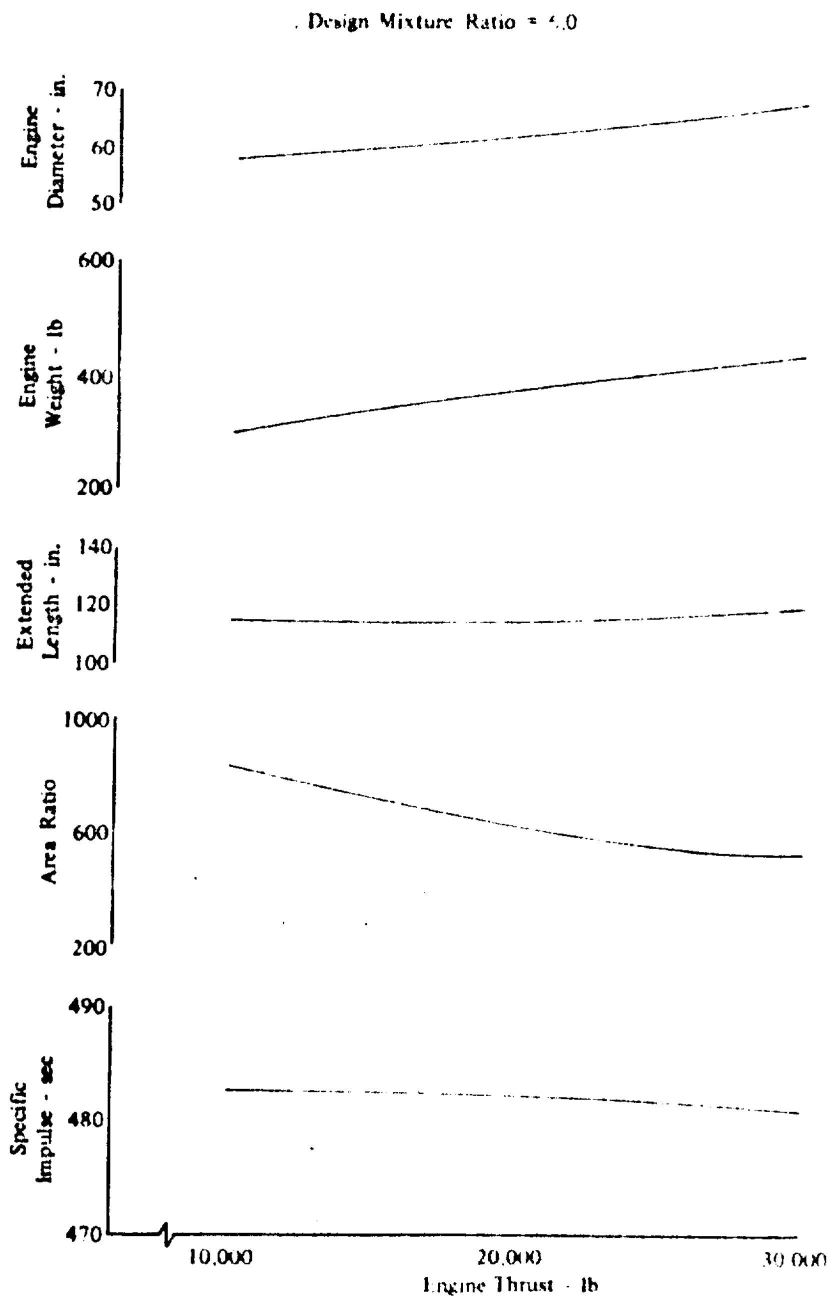

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Figure 3-16. Staged Combustion Engine Parametric Characteristics (60 in Retracted Length)

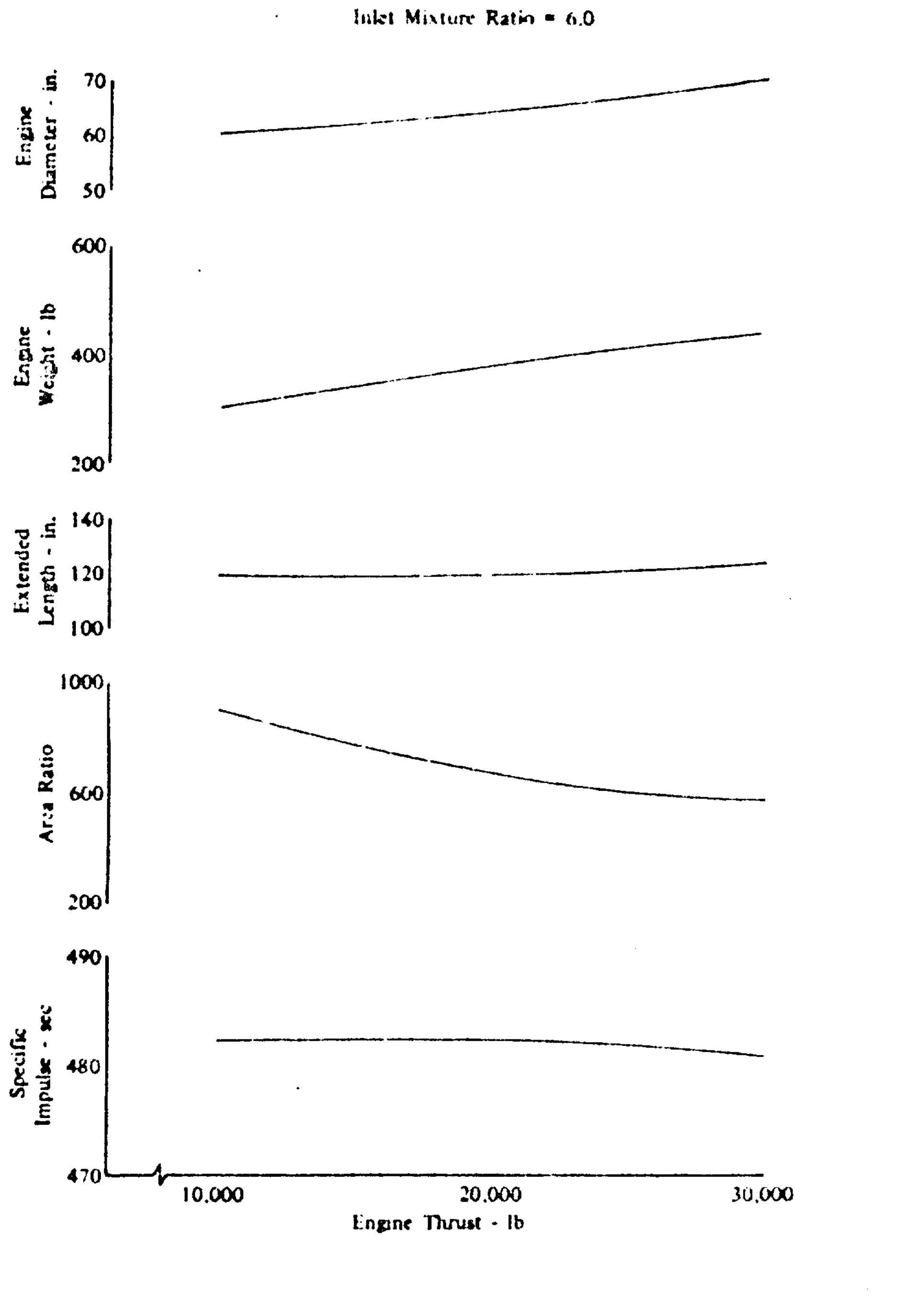

Figure 3-17. Staged Combustion Engine Parametric Characteristics (65 in. Retracted Length)

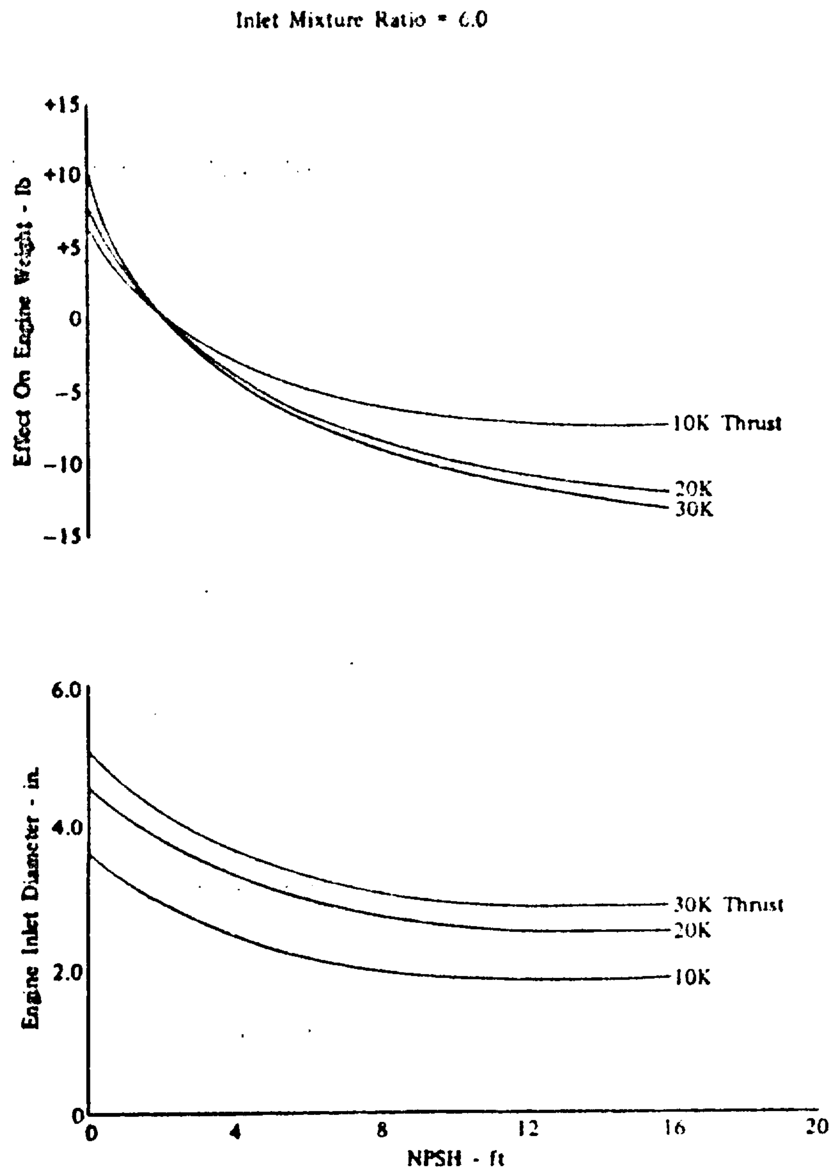

Figure 3-18. Oxidizer Low Pressure Pump NPSH Effects on Engine Weight and Inlet Line Diameter

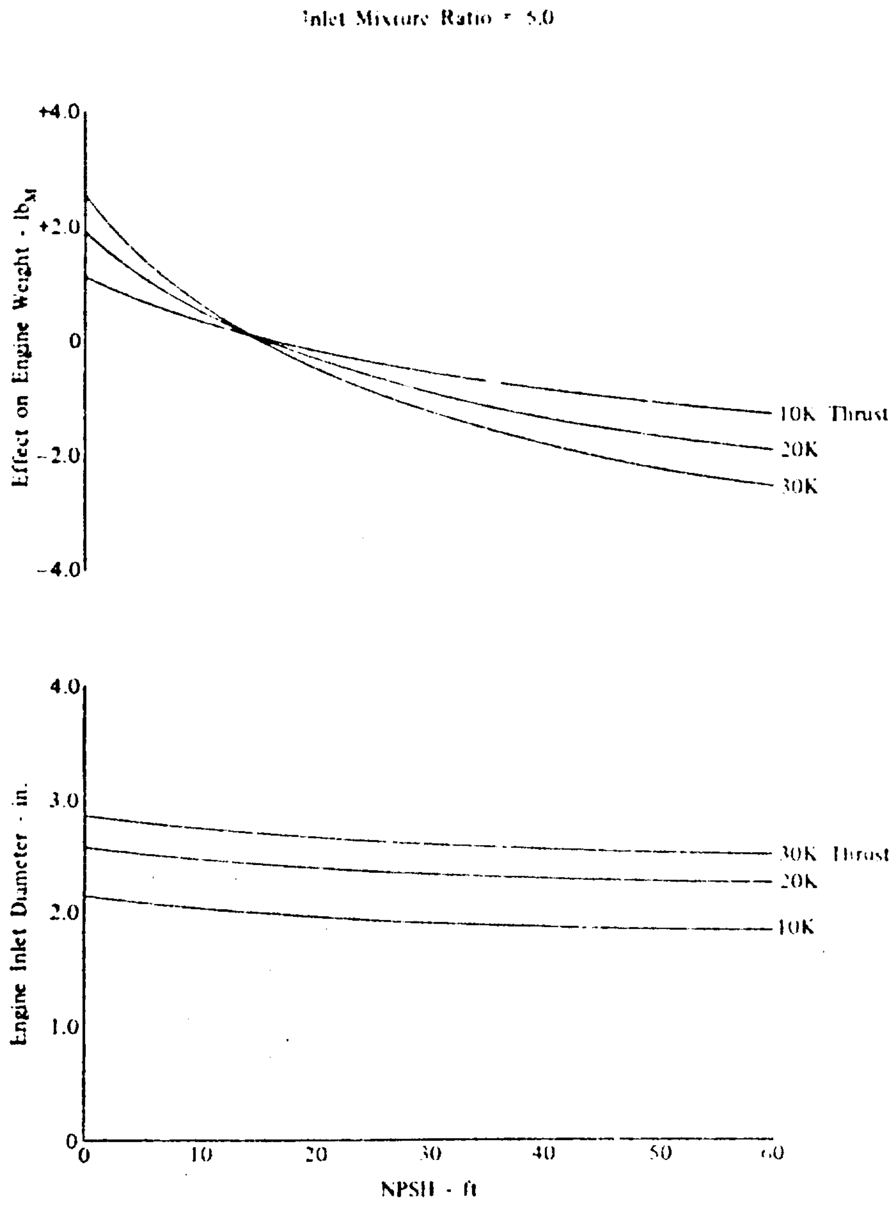

Figure 3-19 Fuel Low Pressure Pump NPSH Effects on Engine Weight and Inlet Line Diameter

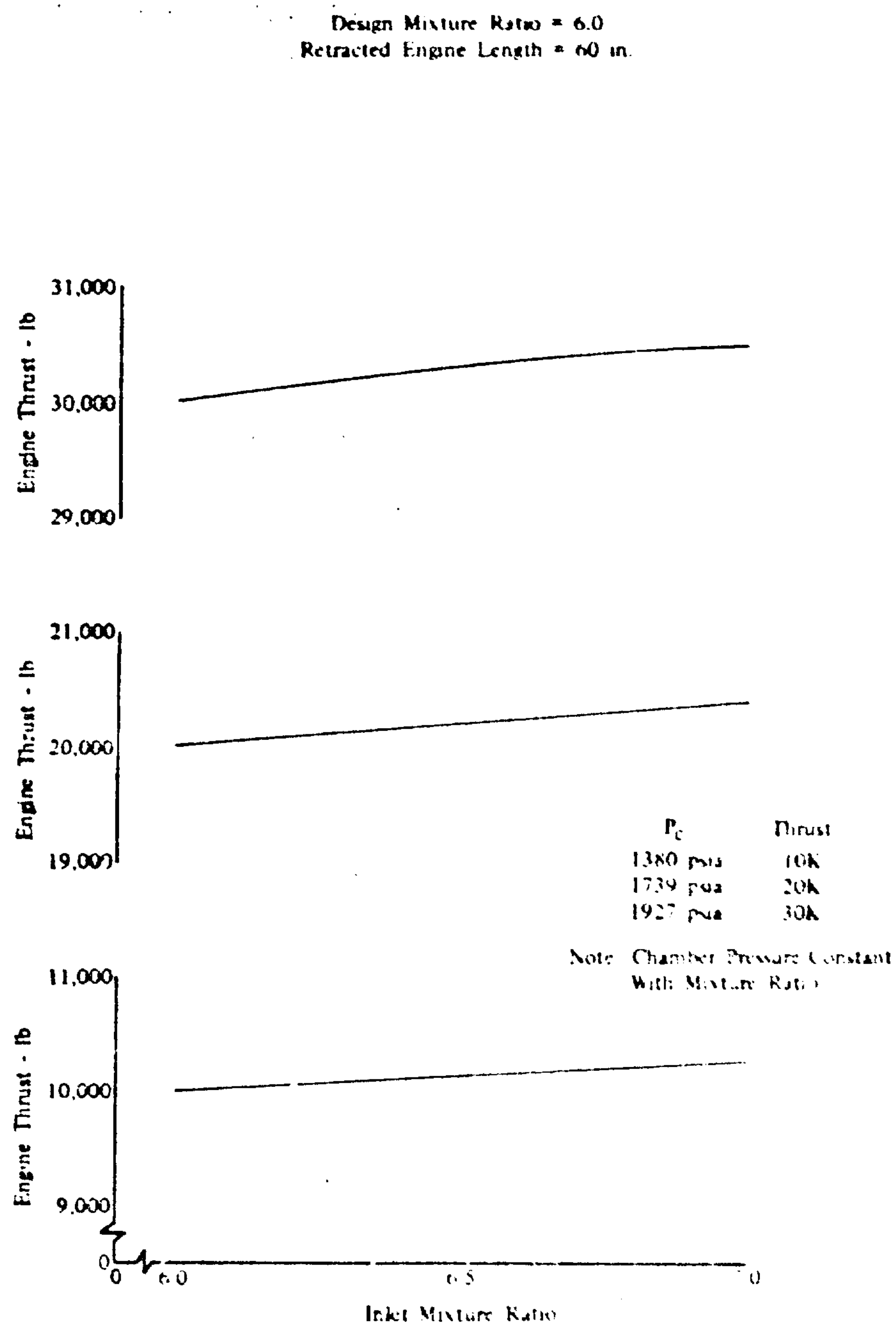

Figure 3-20 Advanced Expander Cycle Off-Design Thrust Characteristics

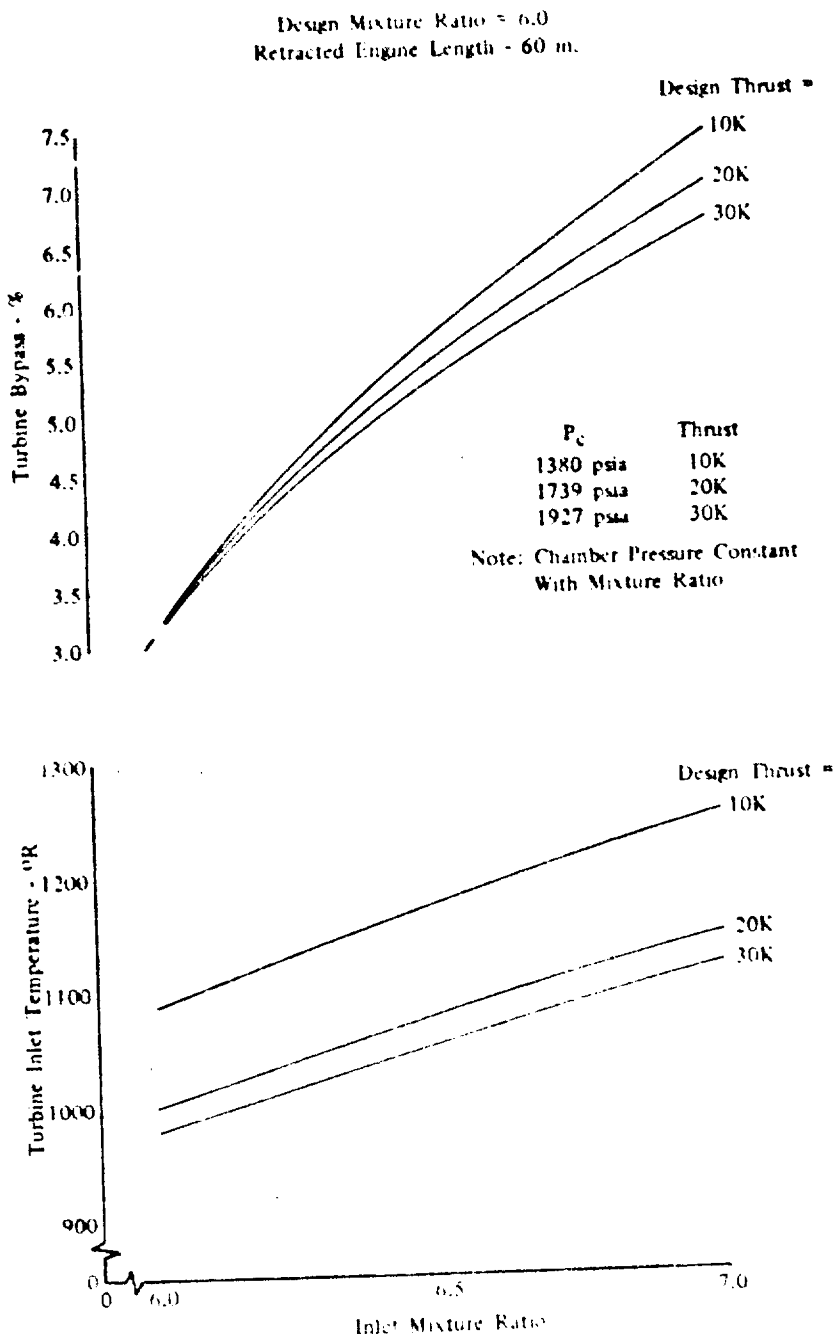

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Figure 3-21 Advanced Expander Cycle Off-Design Turbine Power Characteristics

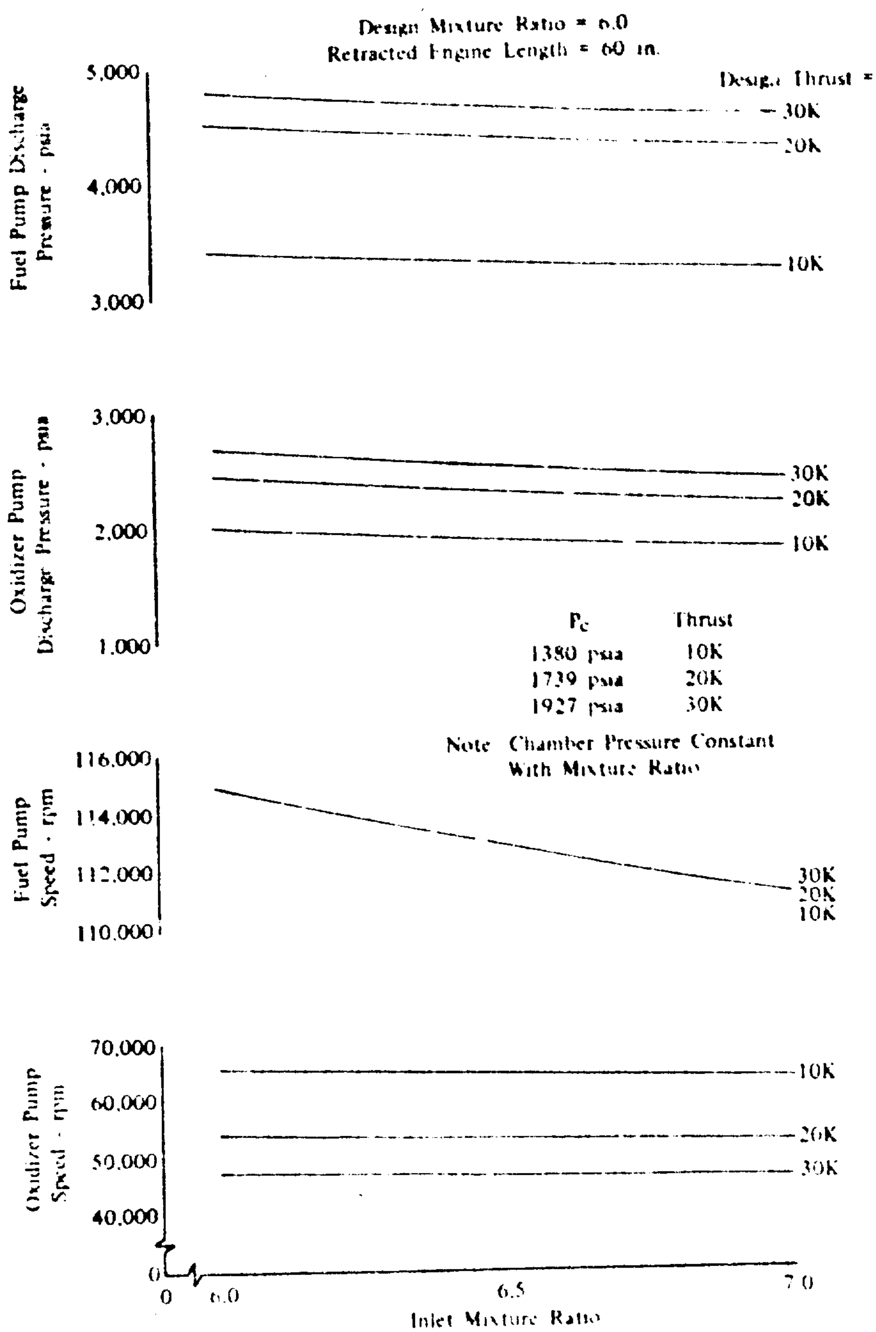

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Figure 3-22 Advanced Expander Cycle Off-Design Pump Characteristics

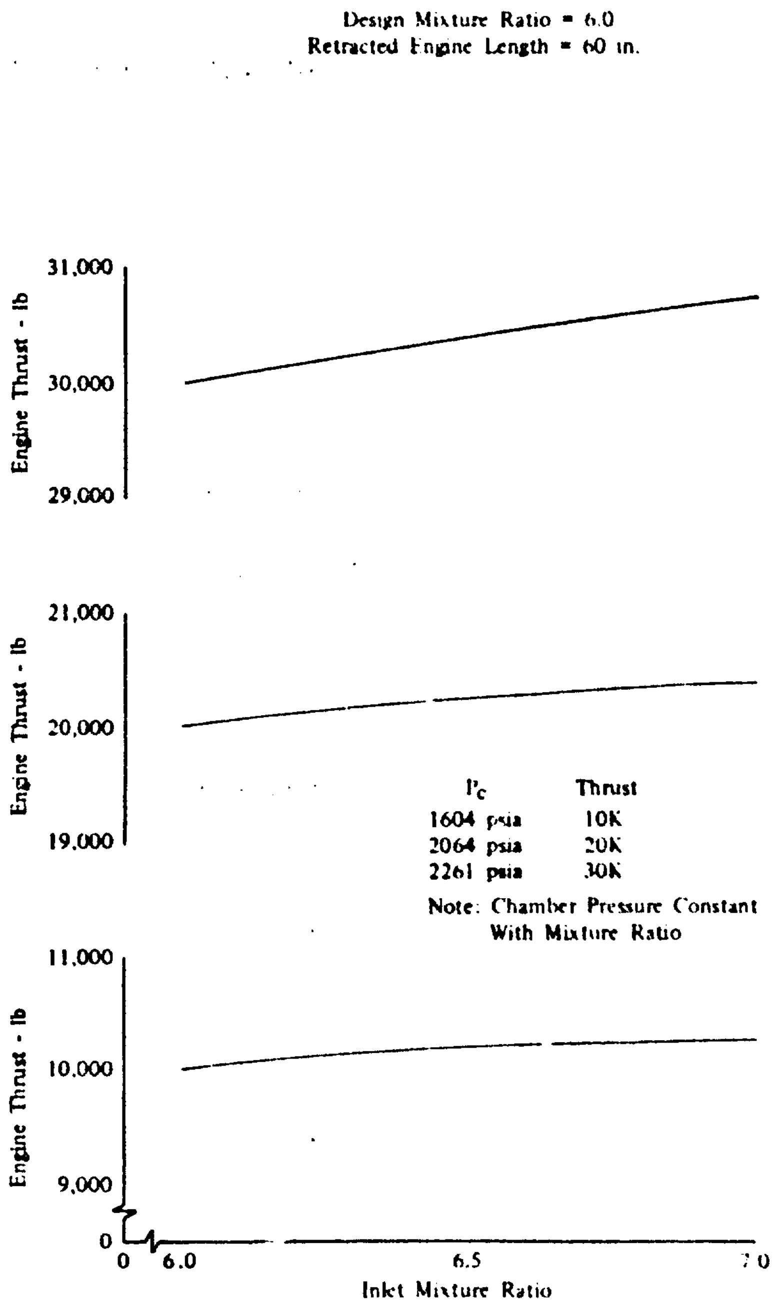

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Figure 3-23. Staged Combustion Engine Off-Design Thrust Characteristics

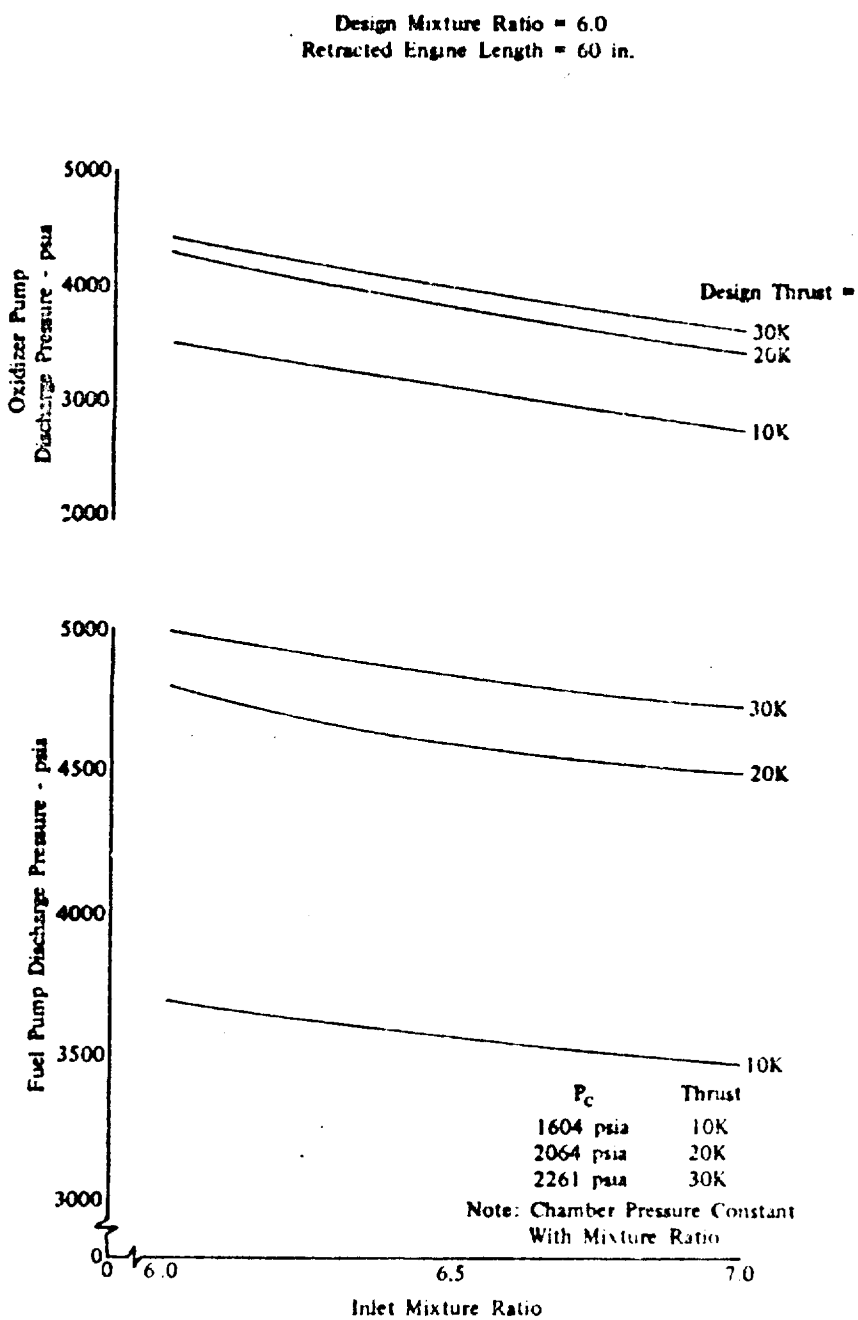

Figure 3-24. Staged Combustion Engine Off-Design Pump Discharge Characteristics

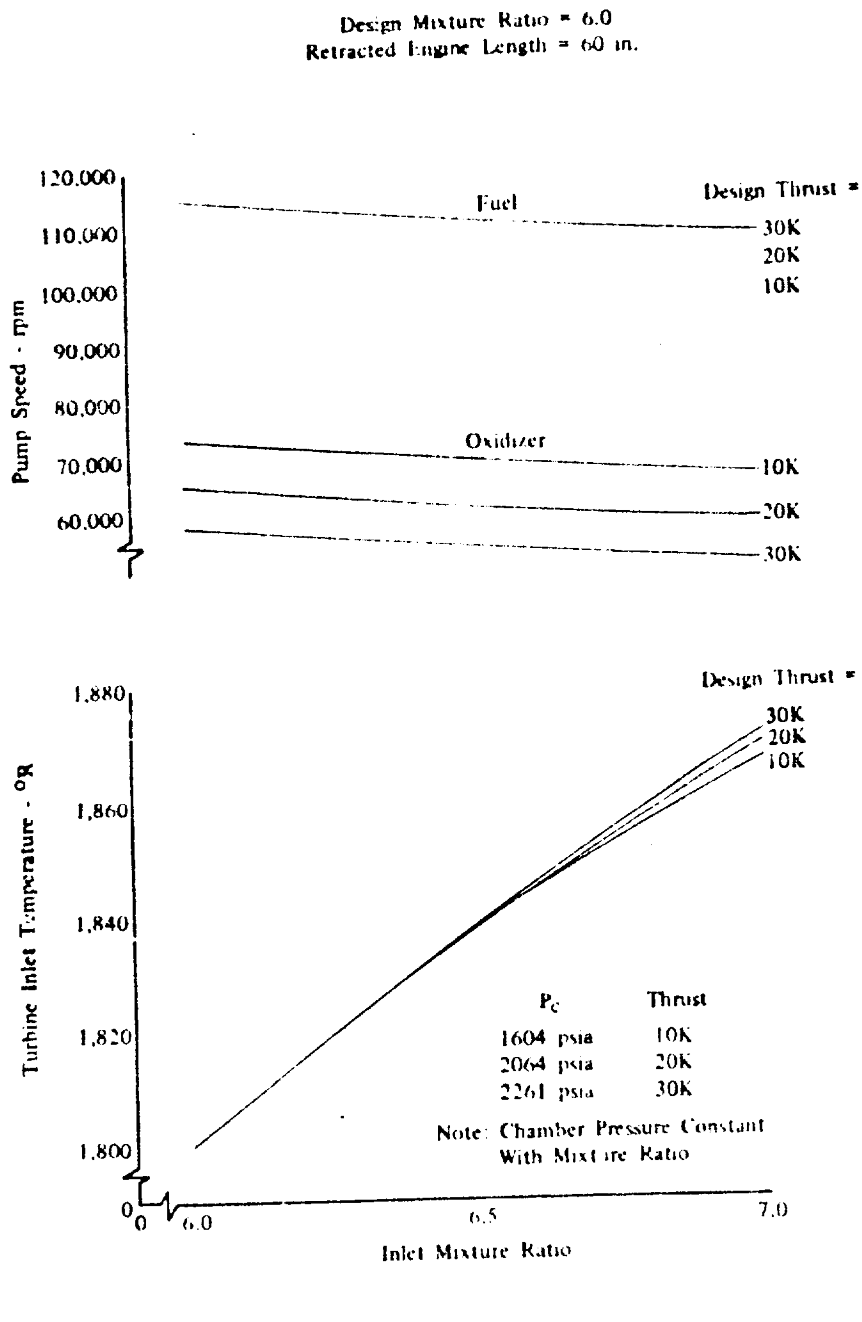

Figure 3-25. Staged Combustion Engine Off-Design Turbopump Characteristics

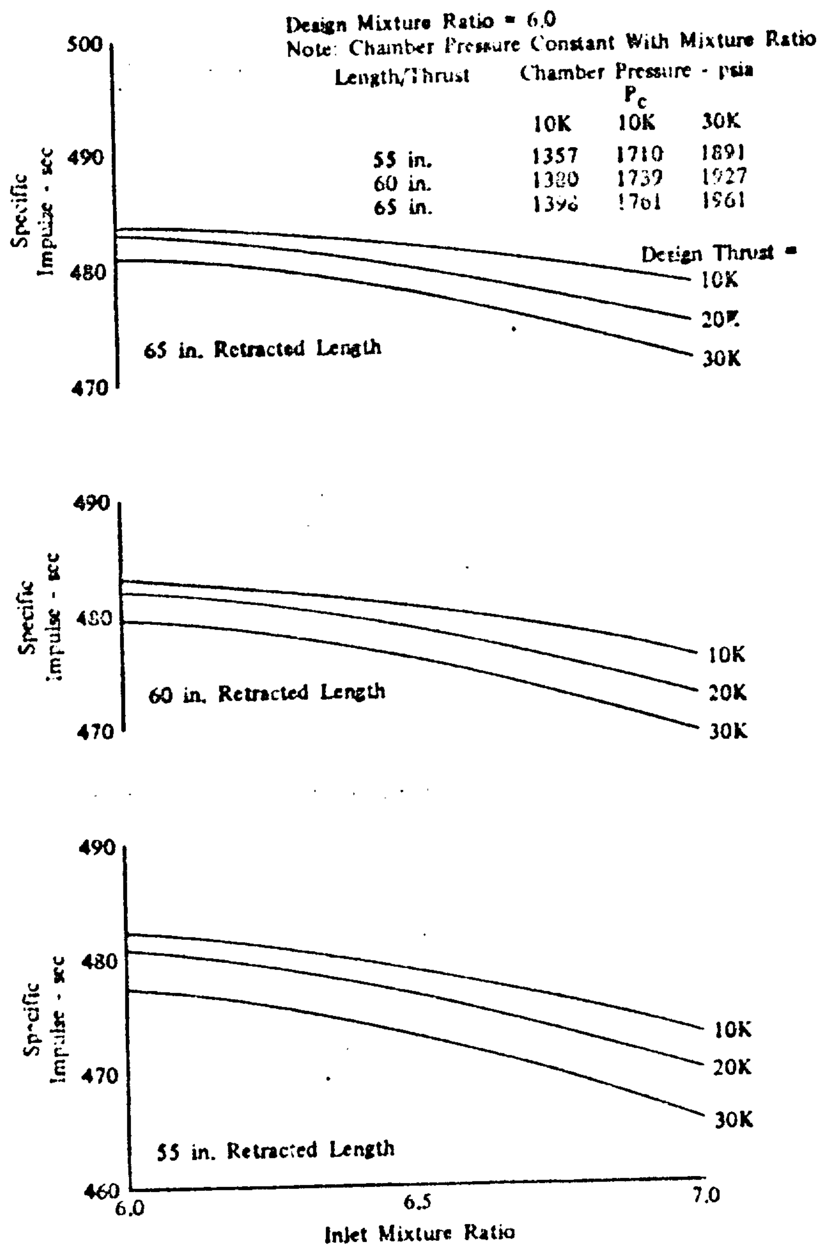

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Figure 3-26. Advanced Expander Cycle Off-Design Performance Characteristics

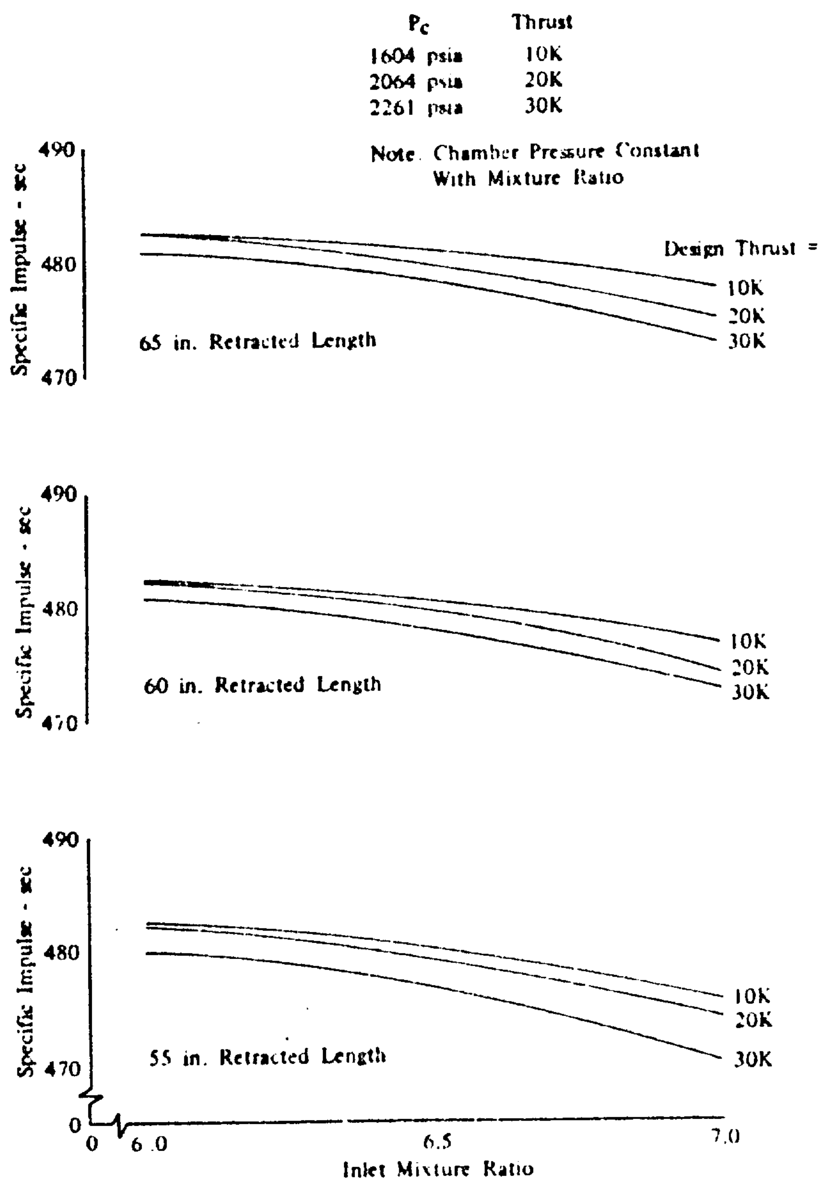

Figure 3-27. Staged Combustion Engine Off-Design Performance Characteristics

3.4.4 Parametric Cost Data

Parametric rough-order-of-magnitude (ROM) cost data were developed for the advanced expander and staged-combustion cycle engines. The advanced expander cycle estimates were generated by applying a factor for a greater "degree-of-difficulty" to the estimates for the RL10 derivative engines whose costs were estimated in 1973 based upon definition of a relatively detailed programatic analysis (see Section 7 of this report), augmented by RL10 program historical data. The staged-combustion cycle estimates were essentially an update of those for the 20,000 lb thrust, staged-combustion engine defined by Pratt & Whitney Aircraft in 1973, (see PWA Report FR-5654, Advanced Space Engine Preliminary Design), which had been estimated based upon a fairly detailed programatic analysis, augmented by experience represented by costs incurred in the P&WA XLR-129 staged-combustion demonstration engine program.

Figures 3-28 through 3-30 present estimated DDT&E, production, and operations program costs (excluding propellant costs) as a function of engine thrust for the advanced expander and staged-combustion cycle engines. These curves represent a "best estimate" of the realistic costs of these programs (i.e., not "success-dependent" nor grossly inflated to cover parallel development efforts in many areas). The figures indicate that engine thrust level has a minimal effect on engine program costs and that a relatively small difference exists in production costs between the two cycles. Operations costs are shown to be essentially the same for the two cycles as they have the same time between overhauls. However, there is a substantial difference in the DDT&E costs of the two cycles. This difference, based on our analysis, is due to the considerably greater complexity of the staged-combustion cycle.

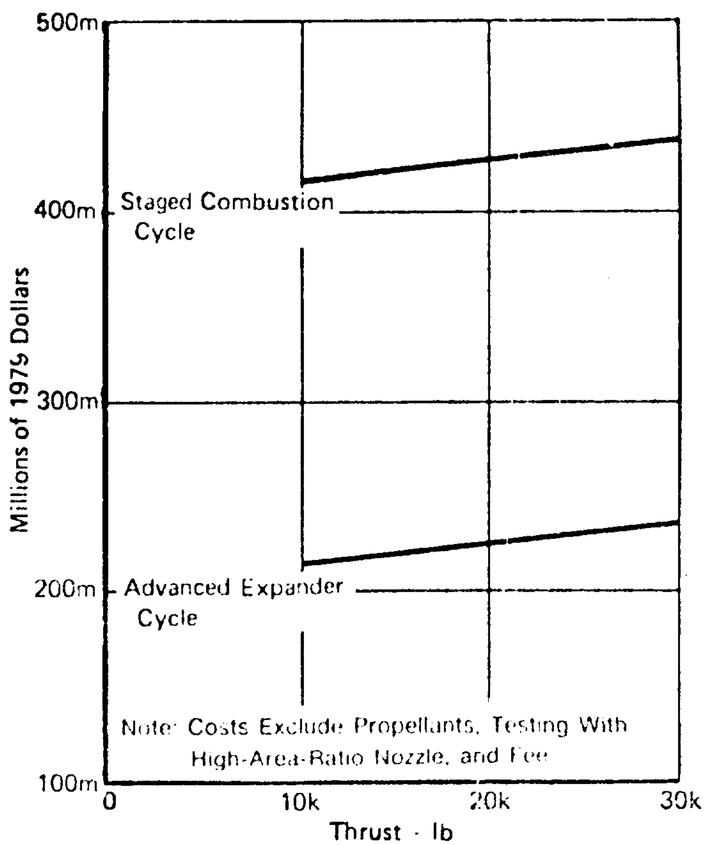

Figure 3-28. OTV Engine Parametric DDT&E Cost

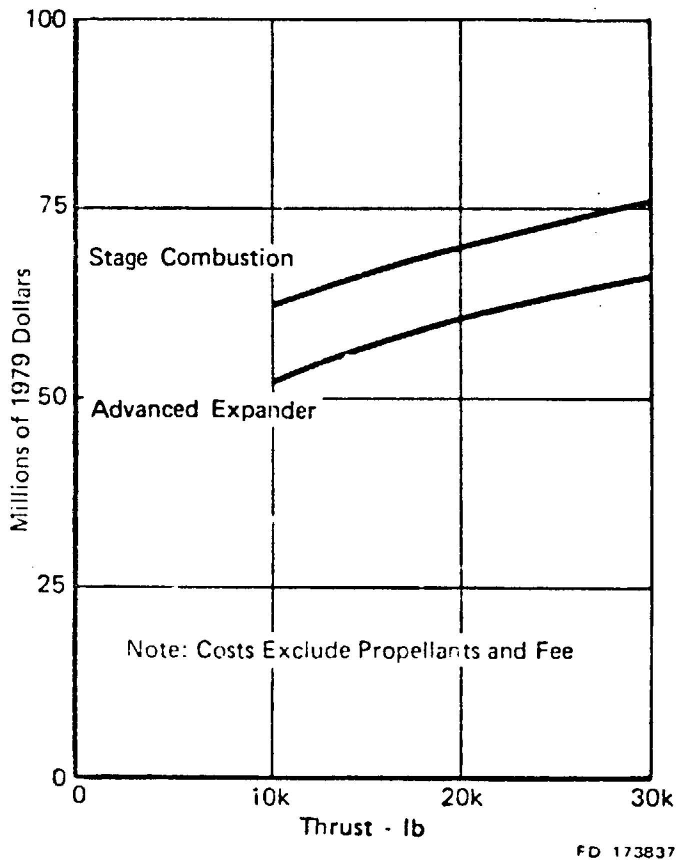

Figure 3-29. OTV Engine Parametric Production Cost, 50 Production Engines

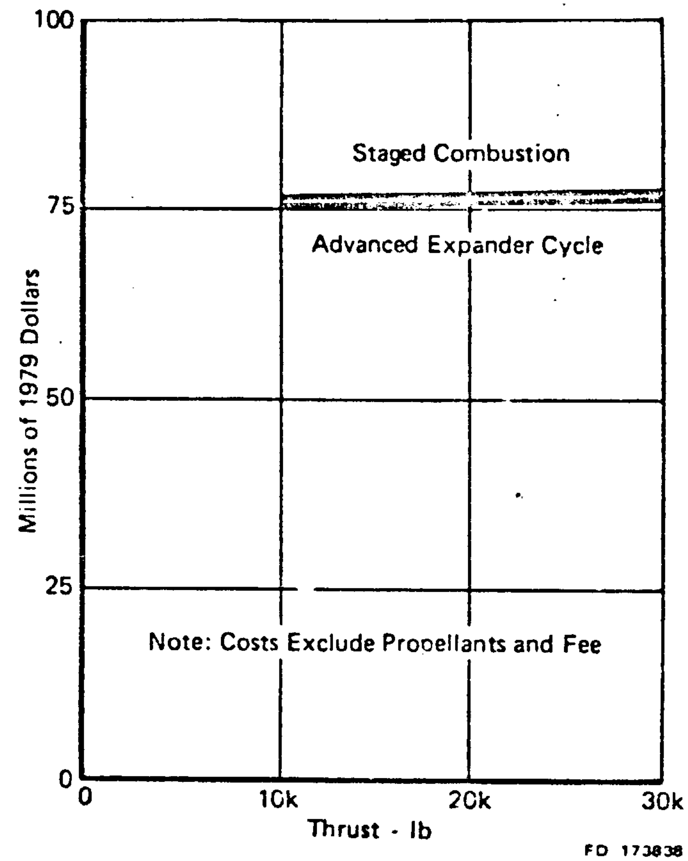

Figure 3-30. OTV Engine Parametric Operations Cost, 15 Flights per Year, 12 Year Program

This increased complexity results in a larger number of component, subsystem, and engine tests as well as an increased amount of engineering design labor and development hardware required to develop the highly reliable, operationally flexible and long life engine required by the OTV. This development cost difference was defined in our 1973 design studies and nothing that has occurred since has indicated that the complexity of a small staged-combustion engine is any lower or that development is less difficult than was indicated at that time.

SECTION 4

ADVANCED-EXPANDER ENGINE OPTIMIZATION

4.0 GENERAL

A prepoint design study was performed to optimize thrust chamber geometry and cooling, engine cycle variations, and controls for an advanced-expande engine. Performance was optimized for advanced-expander engines with thrust levels of 10, 15 and 20K lb at a mixture ratio of 6:1 and an engine retracted length of 60 in. The preliminary cycle studies completed previously provided the starting point for the optimization. The baseline expander cycle configuration was used to optimize the combustion chamber/primary nozzle configuration (chamber length, contraction ratio, coolant passage dimensions, etc). In addition, other studies were conducted to define component variations that might provide performance improvements. The results were evaluated considering performance-weight trade factors, life requirements, impact of control requirements, etc. A preliminary point design engine cycle was determined for each of the three thrust levels and was used to generate power balance points using a bypass flow margin based on RL10 experience.

4.1 COMPONENT OPTIMIZATION

The baseline advanced expander cycle engine configuration (shown in Figure 4-1) was defined in preliminary cycle studies prior to generating the parametric data in Task 2 of this study. Some of the engine configuration selections (such as the geared drive system of the engine boost pumps) were extensively evaluated in earlier rocket engine studies and due to relatively minor effects on engine performance, these items were not considered. The baseline engine configuration provided the starting point for the optimization, and the effects of fuel pump configuration, turbine configuration, regenerator effectiveness and coolant flow routing were evaluated relative to this configuration. Oxidizer system component variations were not evaluated due to their small effect on engine performance. Table 4-1 presents a summary of the results of the cycle optimization; the cycle variations evaluated are discussed separately in the following paragraphs.

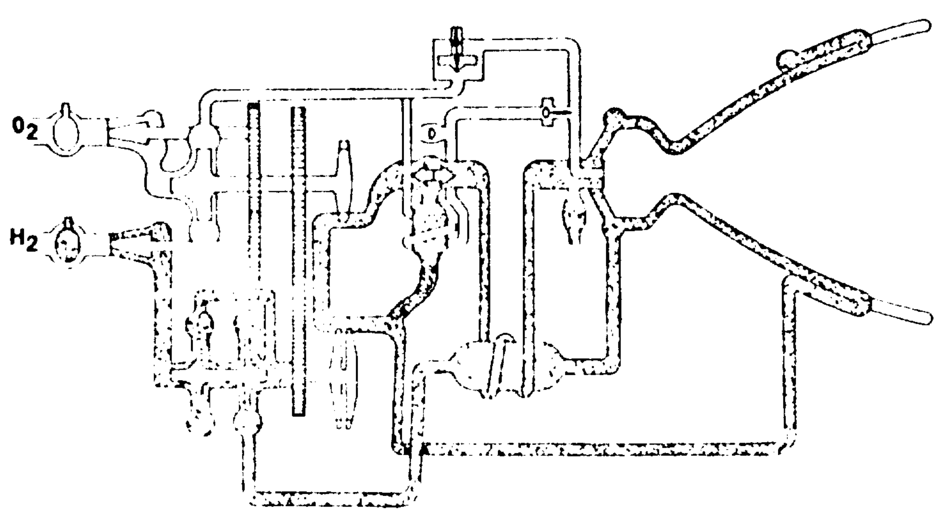

Figure 4-1 Advanced Expander Engine Configuration

TABLE 4-1. ADVANCED EXPANDER ENGINE COMPONENT OPTIMIZATION

Configuration Change from Baseline Engine	*ΔPerformance Effect*	*Comments*
2 Stage to 3 Stage Fuel Pump (10K thrust)	Pc +140 psi Isp +0.9 sec Wt +7 lb Payload +82 lb	Not incorporated because performance increase does not justify added cost and complexity
115,000 to 150,000 Fuel Pump Speed (10K thrust)	Pc +100 psi Isp +0.7 sec Wt 12 lb Payload +110 lb	Not incorporated because performance increase does not justify added cost and complexity
40% to 50% Regenerator Effectiveness (15K thrust)	Pc +140 psi Isp +0.6 sec Wt +11 lb Payload +50 lb	Effectiveness must be limited to keep chamber coolant temperature low enough to meet engine life requirements
Series Turbines to Parallel Turbines (15K thrust)	Pc -25 psi Isp -0.2 sec Wt [illegible] lb Payload -15 lb	Not incorporated because of slightly lower performance and increased flow control complexity
Parallel to Counter Chamber Coolant Flow Routing (15K thrust)	Pc -110 psi Isp -1.5 sec Wt +5 lb Payload -176 lb	Not incorporated because of lower performance

4.1.1 Fuel Pump Configuration

The effects of increasing fuel pump speed from 115,000 to 150,000 rpm and adding a stage to the baseline two-stage pump were evaluated at the 10K thrust level, since the greatest performance effect could be realized there. Figure 4-2 shows the effects on attainable chamber performance of fuel pump speed for both 2- and 3-stage pumps. The +100 psi chamber pressure change realized from the speed increase only increases specific impulse by 0.7 sec, and the 140 psi chamber pressure gain from the additional pump stage is only worth 0.9 sec in impulse. These gains in specific impulse are small because the baseline engine chamber pressure is already ≈ 1500 psia.

Figure 4-3 shows the impulse change resulting from a 100 psi change in chamber pressure as a function of base chamber pressure. The data indicates that the performance gain for a given chamber pressure increase drops significantly as base chamber pressure is increased. The 3-stage fuel pump was not incorporated into the baseline engine configuration because the small performance increase did not justify its greater cost and complexity and because of the possibility of critical speed problems resulting with the longer required pump shaft. The same reasoning applies to the speed increase; the small performance increase does not justify the rise of resulting problems (e.g., seals, shaft diameter, gearing, etc.)

4.1.2 Turbine Configuration

Series and parallel turbine configurations were evaluated. Figures 4-4 and 4-5 compare the chamber pressure attainable with each configuration as a function of fuel pump discharge pressure for 10, 15, and 20K thrust levels. There is less than a 100 psi chamber pressure difference between the configurations at all pump discharge pressure and thrust levels, and the series configuration peaks at a slightly higher chamber pressure. The series configuration was selected for the baseline engine because it requires a less complex flow control than the parallel configuration, and provides essentially equal performance.

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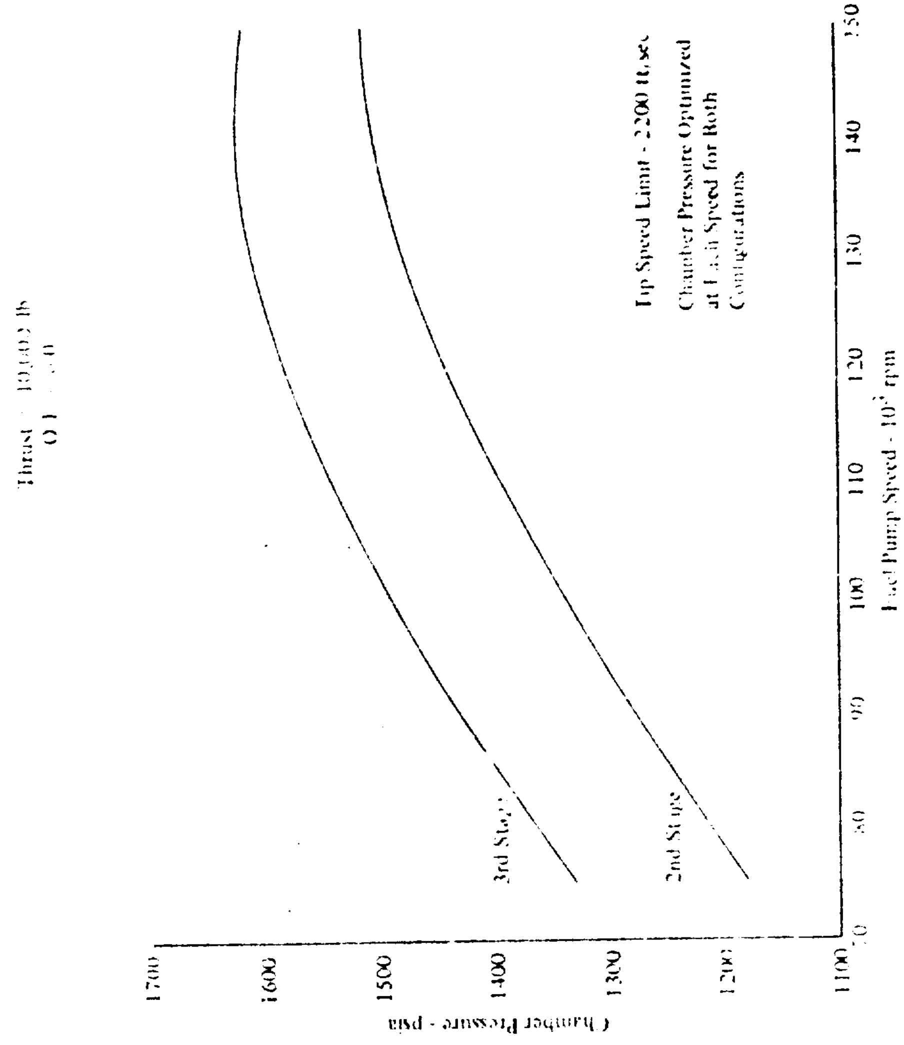

Fuel Pump Configuration Effects on Advanced Expander Engine

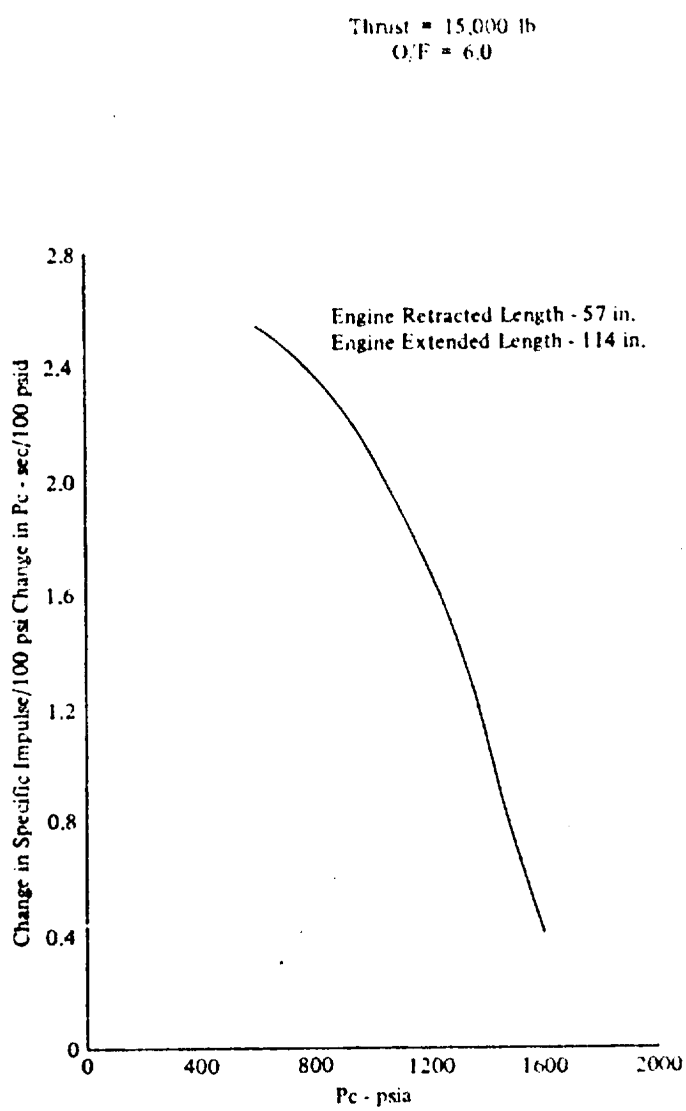

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Figure 4-3. Chamber Pressure Effects on Advanced Expander Engine Performance

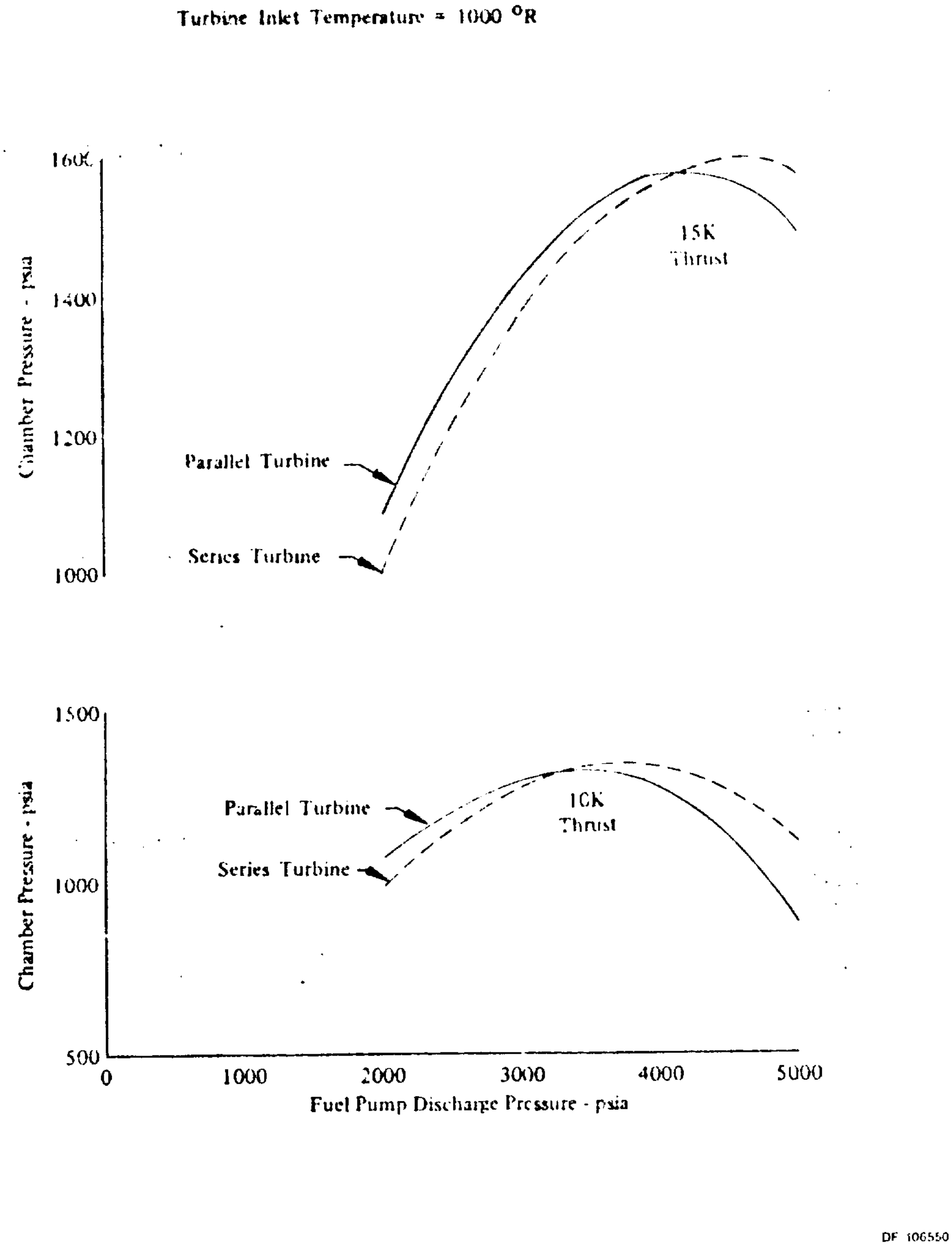

Figure 4-4. Turbine Configuration Effects on Advanced Expander Engine

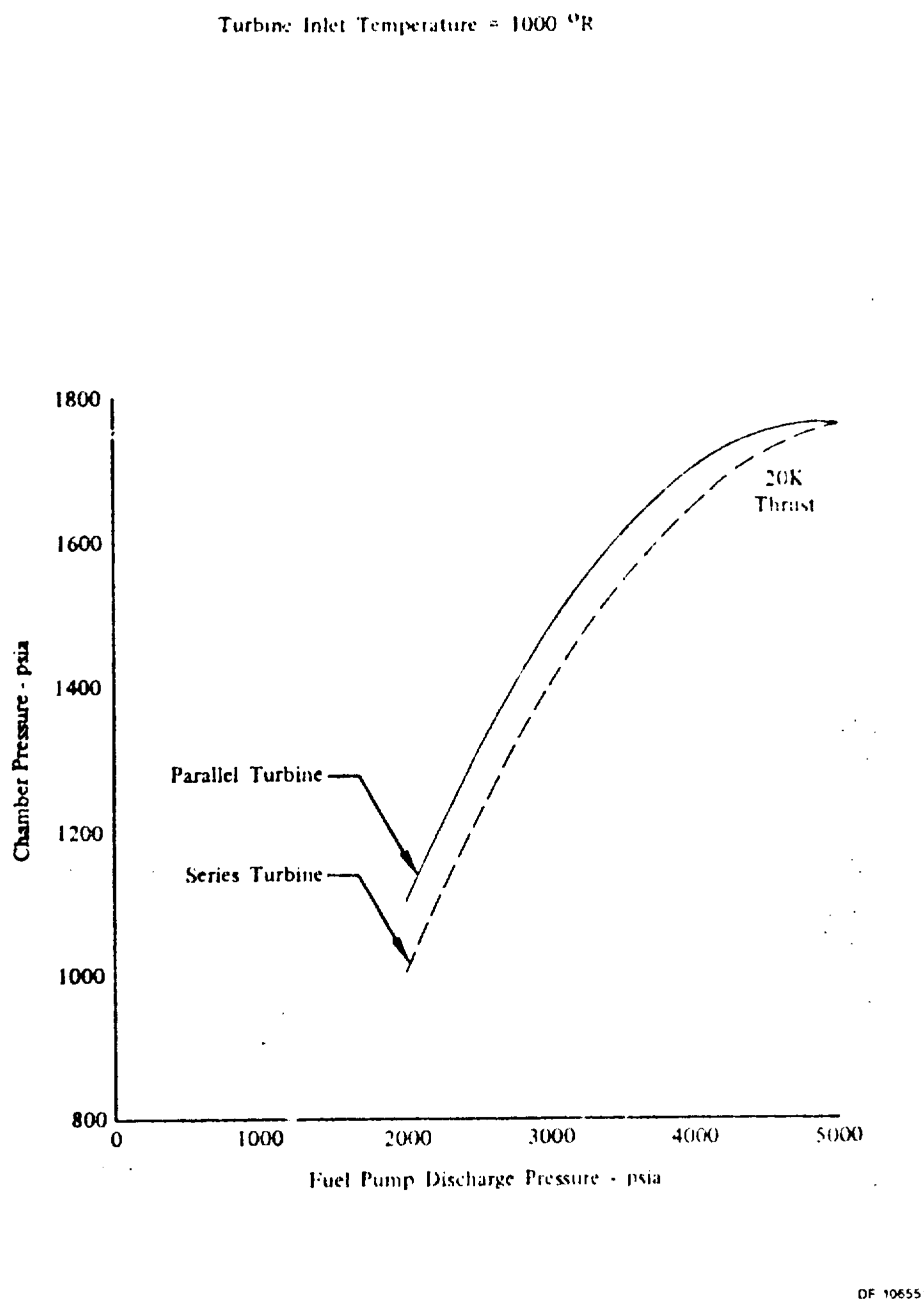

Figure 4-5. Turbine Configuration Effects on Advanced Expander Engine (20K)

4.1.3 Coolant Flow Routing

Parallel and counterflow chamber/nozzle coolant flow routing schemes were evaluated (see Figure 4-6). Split flow cooling was not evaluated because it provides no performance advantage for a system with geared engine-mounted boost pumps, and requires a positive control to provide the desired flow split at all operating conditions. Parallel flow cooling is not normally used for a regeneratively cooled chamber/nozzle because the high heat-flux levels in the chamber would cause significant density changes (~8 to 1) in the hydrogen coolant introduced at pump discharge conditions. This could result in insufficient cooling during off-design engine operation increasing the risk of coolant channel burnout. A regenerator, as used in the baseline advanced expander cycle engine, heats the hydrogen prior to routing to the thrust chamber coolant channels limiting density changes to less than 2 to 1 in the channels. This allows the use of either parallel or counterflow cooling. Figure 4-7 presents a comparison of parallel and counterflow configuration results. A chamber length limit of 15 in. was determined for the parallel configuration based on cooling limits at the throat. An 18-in. counterflow configuration was then evaluated to determine if any advantage could be achieved. Coolant temperature rise characteristics are about the same for both configurations. Pressure losses are much higher for the counterflow configuration due to greater manifolding losses. For the parallel configuration, coolant Mach number is reduced downstream of the throat due to the nozzle expansion. In the counterflow configuration, the coolant exit manifold is at the injector where relatively high Mach number levels are required to provide adequate cooling, resulting in excessive pressure losses. The parallel cooling configuration was therefore selected.

4.1.4 Regenerator Effectiveness

The effect of regenerator effectiveness on engine performance was evaluated. Figure 4-8 shows payload, weight, and specific impulse characteristics as functions of regenerator effectiveness. Engine performance is improved as effectiveness is increased. However, as effectiveness is increased, the combustion chamber coolant inlet temperature is also increased. As the coolant temperature is increased, a limit is reached where the combustor hot-sidewall temperature can no longer be maintained at levels that meet engine life requirements. The limits on regenerator effectiveness to maintain engine life were determined to be 36, 43, and 48.5% for the 10, 15, and 20K thrust levels, respectively, with a chamber length of 15 in. and chamber contraction ratio of 4.

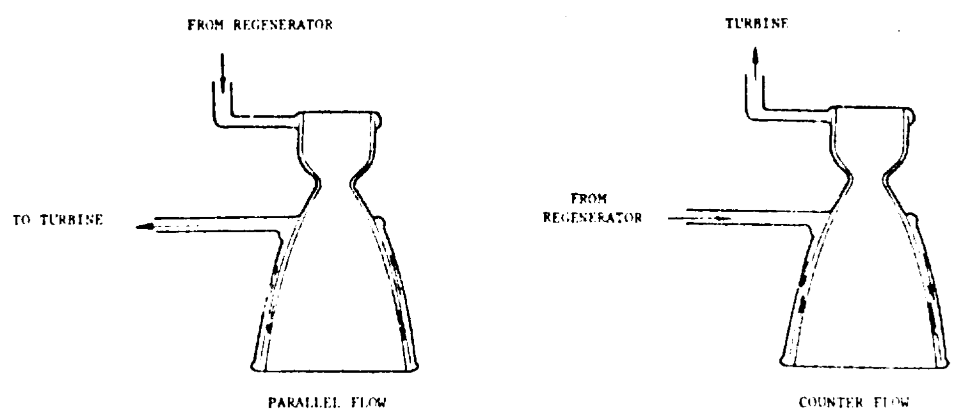

Figure 4-6. Parallel and Counterflow Chamber/Nozzle Coolant Flow Routing Schemes

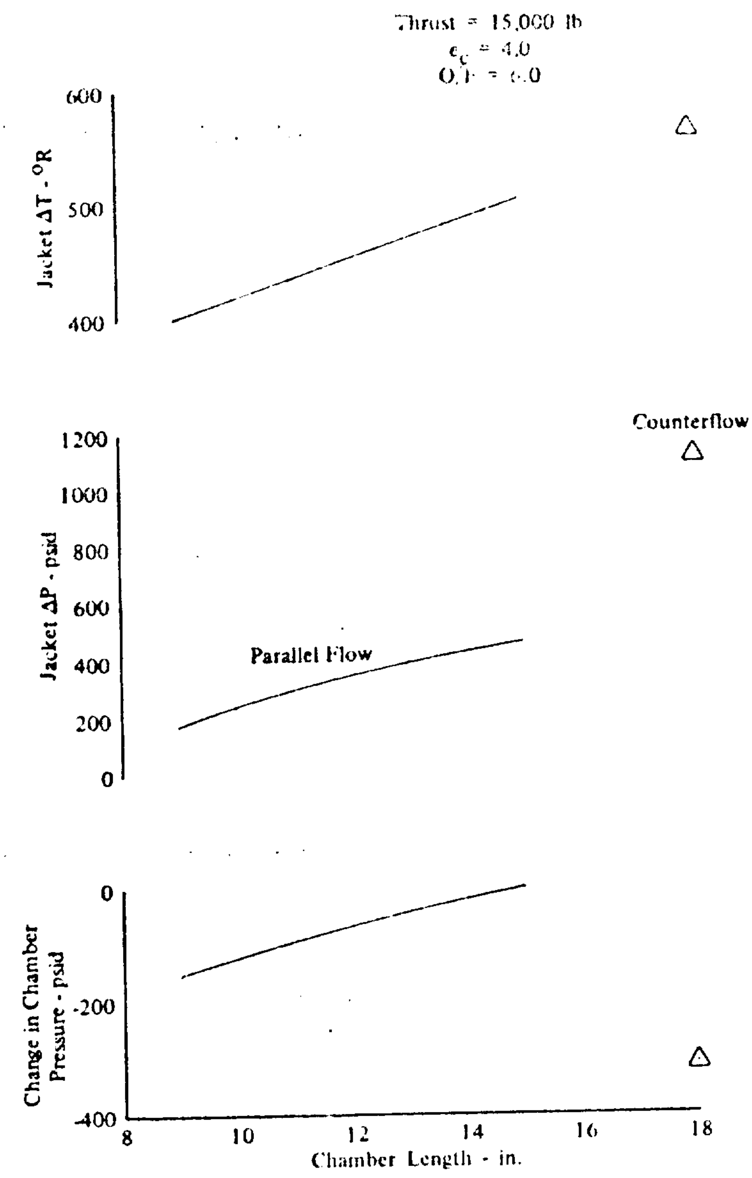

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Figure 4-7. Chamber/Nozzle Coolant Flow Routing Performance Comparison

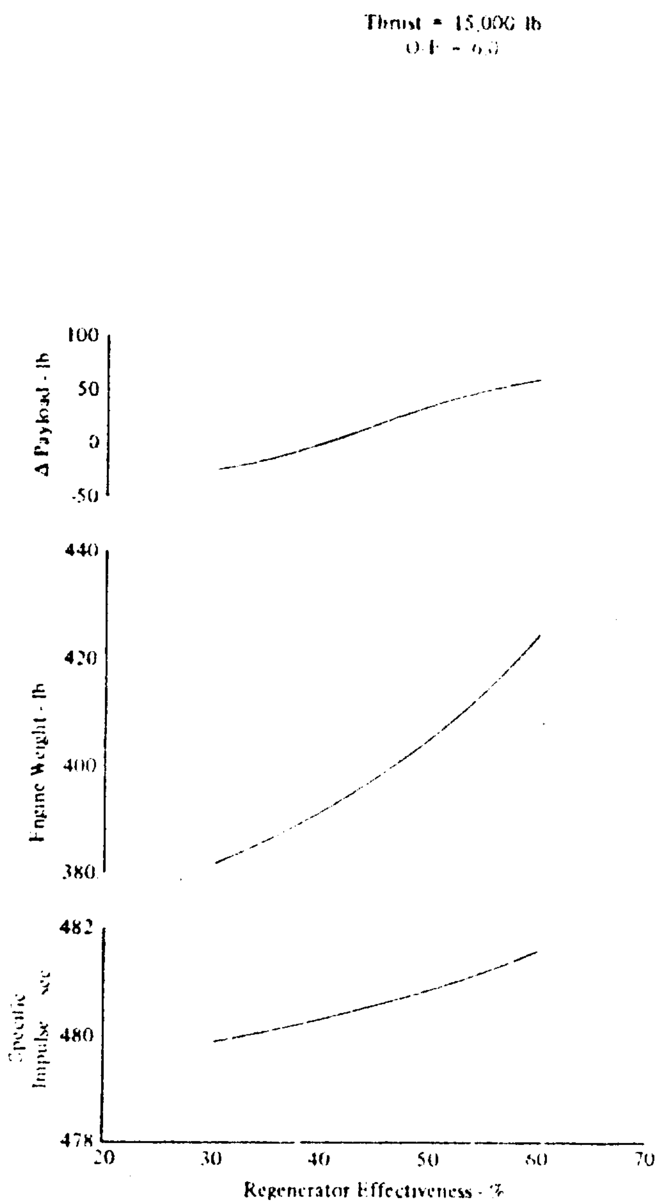

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Figure 4-8. Regenerator Effectiveness Effect on Advanced Expander Engine Performance

4.2 COMBUSTION CHAMBER/PRIMARY NOZZLE OPTIMIZATION

Chamber length and chamber contraction ratio effects on engine performance and life were evaluated at steady-state conditions. Thrust chamber coolant temperature rise, coolant pressure loss, and wall temperature characteristics were predicted. The results were used to update engine cycle performance predictions and assess chamber cycle life. The optimization was completed at a thrust level of 15K lb, and the results were extrapolated to the 10 and 20K lb thrust levels. A chamber length of 15 in. and a chamber contraction ratio of 4 were selected as optimum for the advanced expander cycle engine.

4.2.1 Chamber Length and Contraction Ratio

Chamber length effects on coolant temperature rise and pressure loss were evaluated over a range of 9 to 24 in. at a contraction ratio of 4 and a thrust level of 15K lb. The maximum chamber length that satisfied ground rule requirements were determined to be 15 in.

Contraction ratio effects were then evaluated at chamber lengths of 9, 12, and 15 in. Figure 4-9 shows the effects of chamber length and chamber contraction ratio on coolant temperature rise, pressure loss, and chamber pressure. Figure 4-10 shows the effects on specific impulse, weight, and payload. The 15-in. chamber length with a contraction ratio 4 configuration showed the best performance. Thrust points of 10 and 20K were then evaluated with the 15-in. chamber length and contraction ratio 4 configuration, and this configuration was identified as near optimum. Preliminary design points were generated for all three thrust levels with the 15/4 configuration. Detailed thermal maps (see Figure 4-11) were generated at critical locations to provide the necessary information for the life analysis.

4.2.2 Cycle Life

The cyclic life-limiting engine component is the nontubular portion of the chamber assembly. Large thermal gradients are generated in a regeneratively cooled thrust chamber during operation. These gradients affect the expected cyclic life of the chamber walls, as does the further aggravating condition of high-pressure coolant within the wall passages. Pratt & Whitney Aircraft has developed a method for analysis of nontubular thrust chambers that provides a complete evaluation of thermal cycle life, in that thermal gradients and the pressure differential across the hot wall are simultaneously treated. The plastic strains predicted by this procedure are used with NASA-Lewis strain range/LCF curves to predict cycle life. The cycle life of the nontubular portion of the Advanced Expander Engine combustion chamber has been estimated for the 10, 15, and 20K-lb thrust design points, and the life characteristics satisfy the design requirement level of 1200 cycles (i.e., 4 times the 300 engine duty cycles), as shown in Figure 4-12.

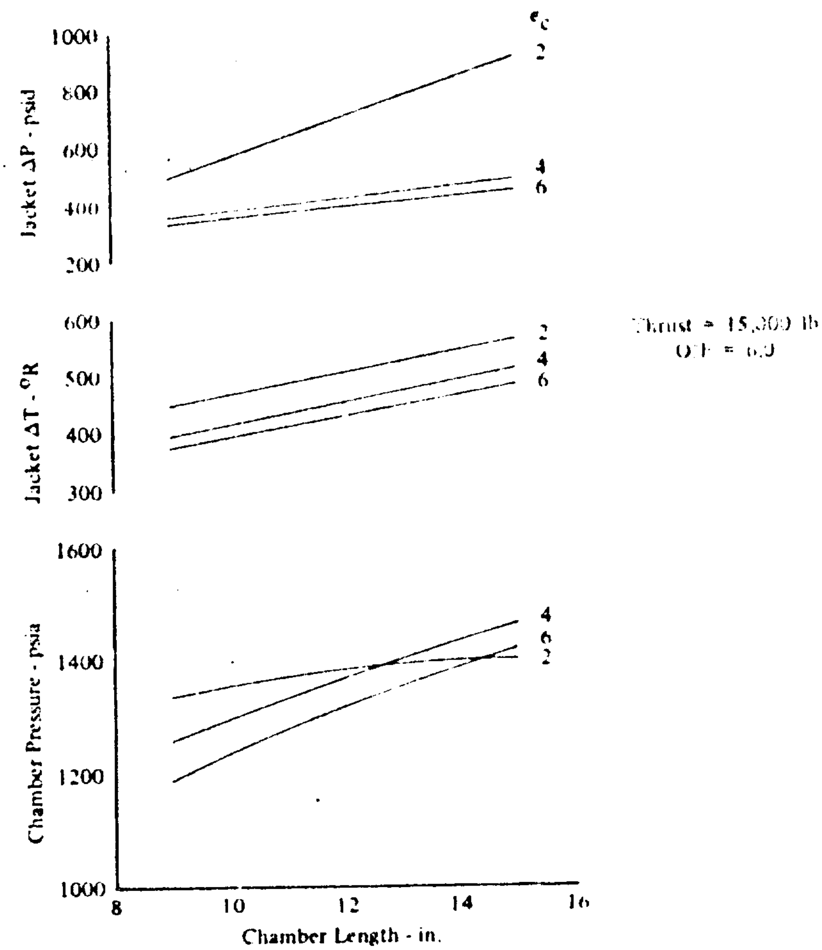

Figure 4-9. Chamber Configuration Effects on Advanced Expander Engine Cycle

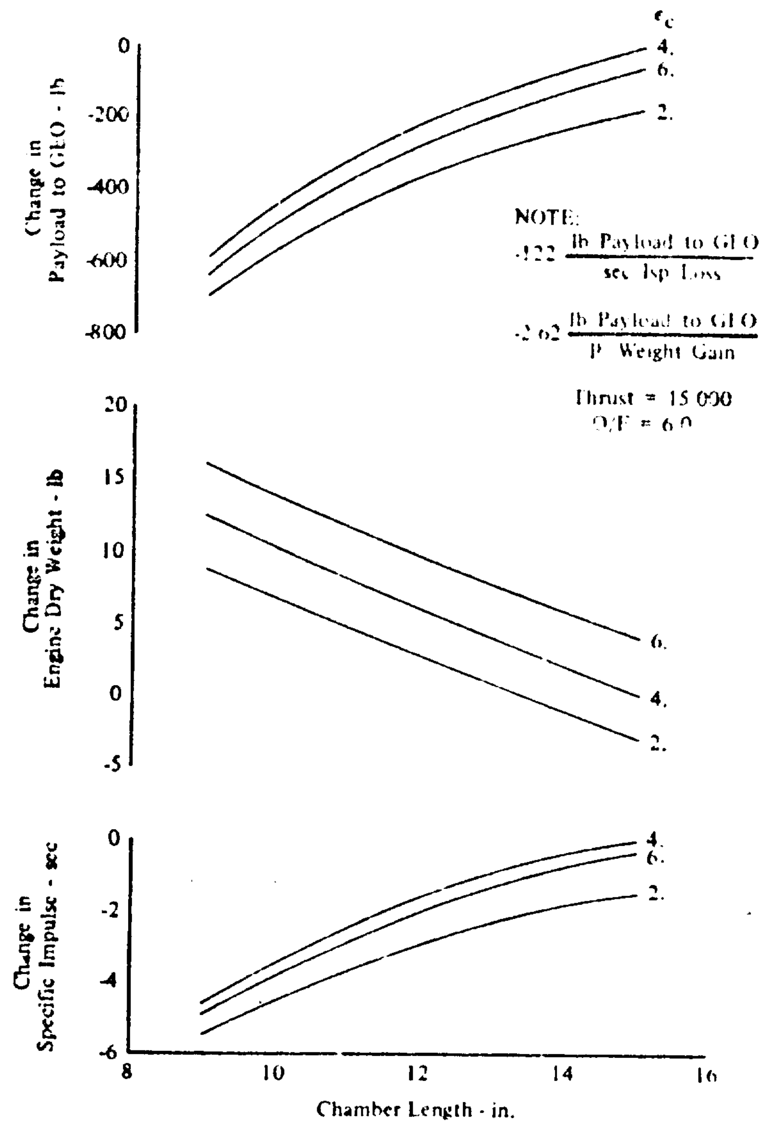

Figure 4-10. Chamber Configuration Effects on Advanced Expander Engine Performance

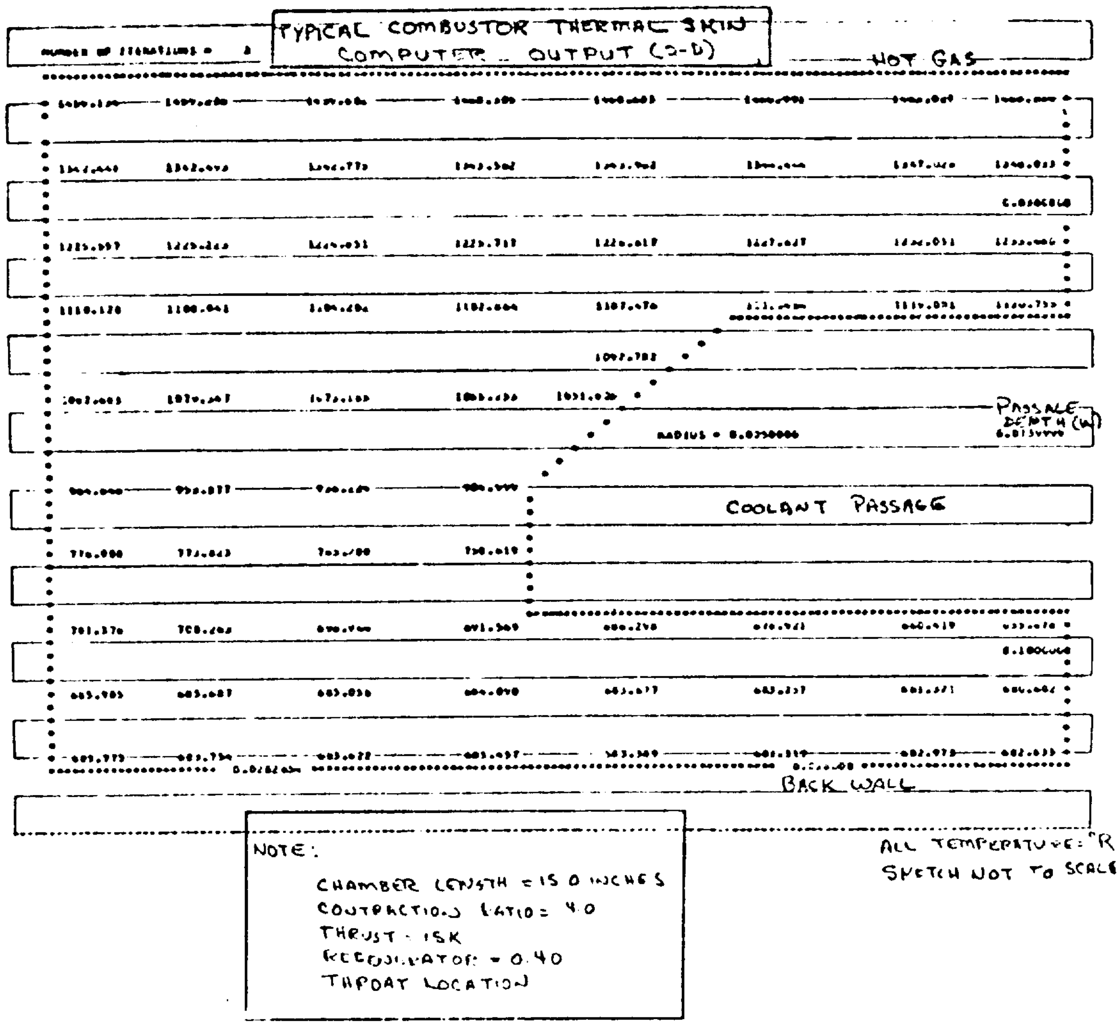

Figure 4-11. Detailed Thermal Map

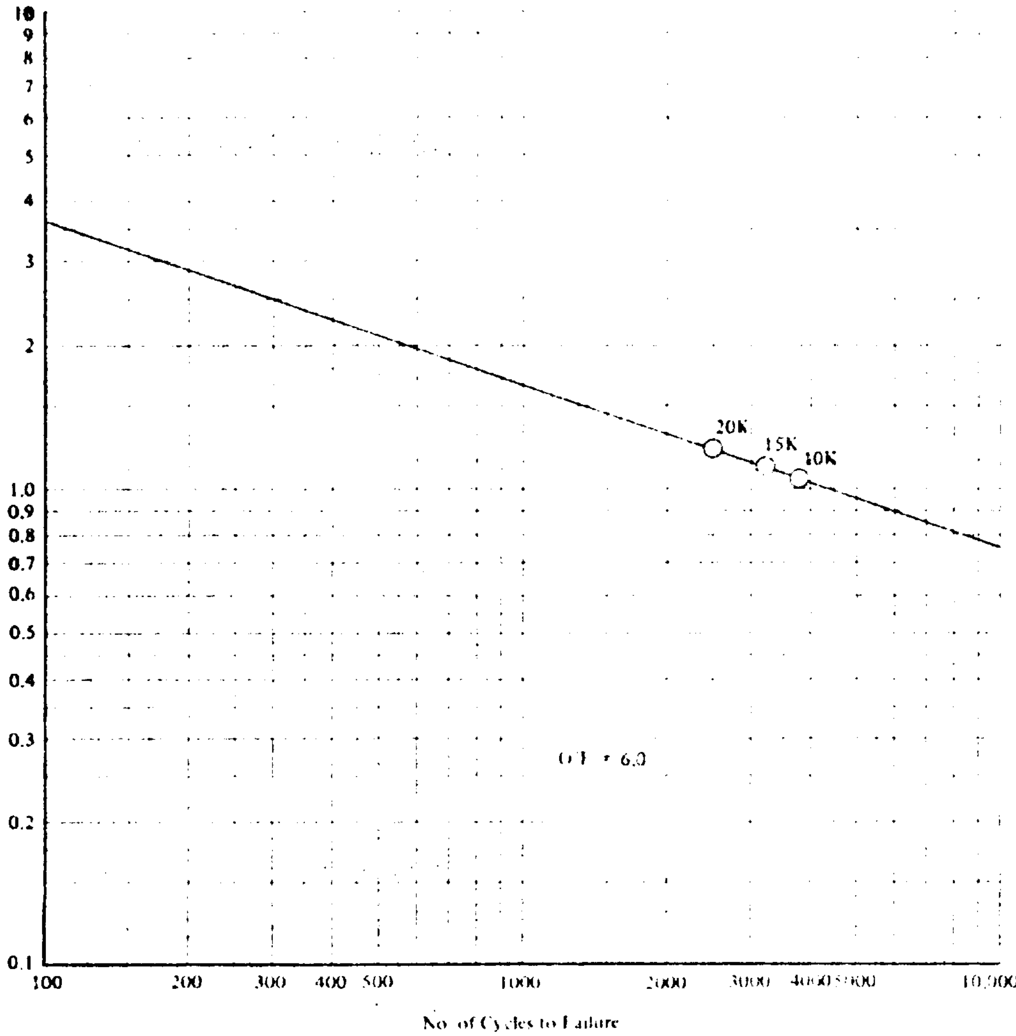

Figure 4-12. Advanced Expander Cycle Engine Design Point Cycle Life Characteristics

4.3 ENGINE OPTIMIZATION

The results of the cycle optimization and chamber/nozzle optimization were evaluated, and preliminary point designs were generated for 10-, 15-, and 20K-lb thrust levels. These point designs meet the engine configuration and life requirements specified in the SOW. Control requirements were evaluated, and a passive control configuration was selected based on the higher cost and lower reliability of a minimum active control configuration. A power margin requirement of 3% turbine bypass flow was established based upon RL10 production engine data. Engine thermal conditioning/start losses were evaluated, and a tank head idle mode of engine operation was selected to minimize losses during thermal conditioning and start.

4.3.1 Engine Power Margin

RL10A-3-3 production engine data was surveyed to determine the expected variation in engine-to-engine performance. Eighteen production engines had been run during acceptance testing with the necessary measurements to determine bypass flow. Percent bypass flow is shown in Figure 4-13 for each of these production engines. Two standard deviations (95% confidence level) of bypass flow were determined to be 2.93%. Accordingly, turbine bypass flow (for the advanced expander cycle engine) was set at 3% to provide margin for statistical deviations from nominal component operating characteristics.

4.3.2 Control Requirements

Passive control and minimum active control configurations were evaluated for the advanced expander cycle engine. Control system simplicity is a primary concern in a new engine design. Exotic closed-loop controls are highly expensive, relatively unreliable, and require substantial development testing. The advanced expander cycle engine valve requirements are similar to those of the RL10 Category IV engine (shown in Figure 4-14), therefore, similar valve configurations were considered in this study. As many of these valves as possible will be actuated using internal engine pressures. Valves actuated by engine-supplied pressures do not require coordination of external pneumatic and electric supplies and tend to normalize the engine transients.

All of the valves operate in an open-loop mode for a passive control configuration. In a minimum active-control configuration (RL10 Category IV type) a thrust control is added to the main fuel control to maintain constant chamber pressure at rated thrust. The thrust control is a normally closed, servo-operated, variable-position bypass valve used to control chamber pressure by the regulation of turbine power (i.e., the control senses chamber pressure and varies turbine flow to maintain a constant pressure).

The effects of engine inlet pressure variations on thrust, mixture ratio, and impulse are shown in Figure 4-15 for a passive control configuration. Less than a 2% change in thrust would be expected during a long burn due to inlet pressure decay (assuming no tank repressurization) with a passive control.

Figure 4-16 shows off-design mixture ratio thrust characteristics for both control configurations. The passive control gives a 10% increase in thrust when operated off-design at a mixture ratio of 7, and the minimum active control configuration gives a 2% increase. This increase would only occur if an active propellant utilization system was required by the OTV. Otherwise, if off design mixture ratio was required for a specific mission, the engine could be trimmed (fuel control and oxidizer controls adjusted) to provide nominal thrust at the desired mixture ratio.

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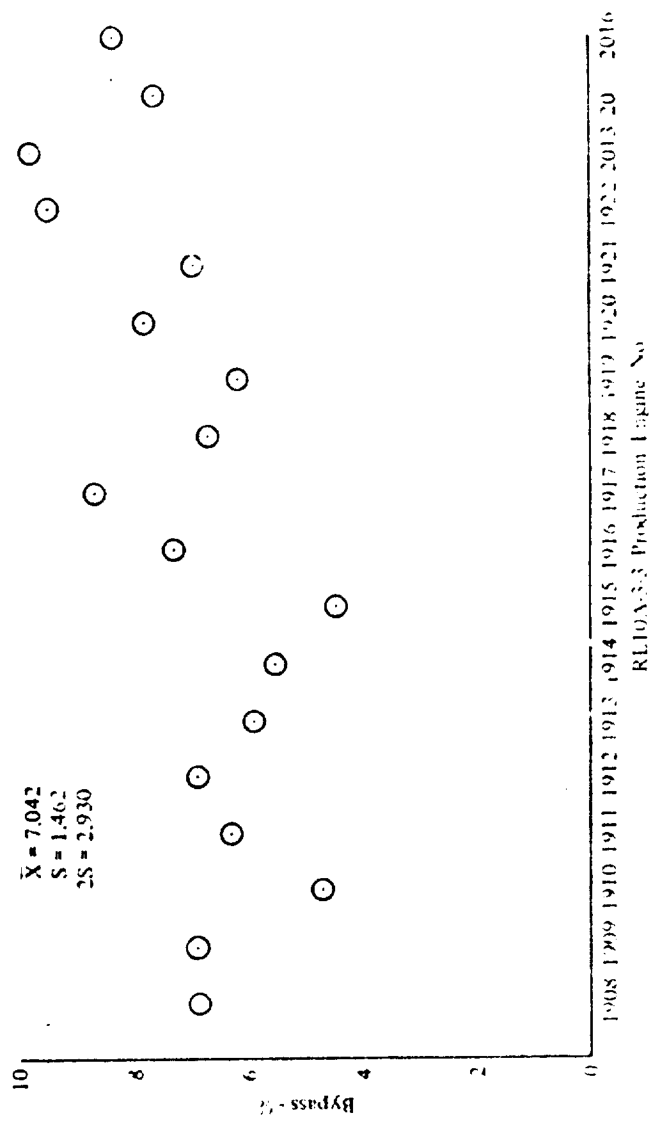

Figure 1-13 RL10 Turbine Bypass Flow Characteristics

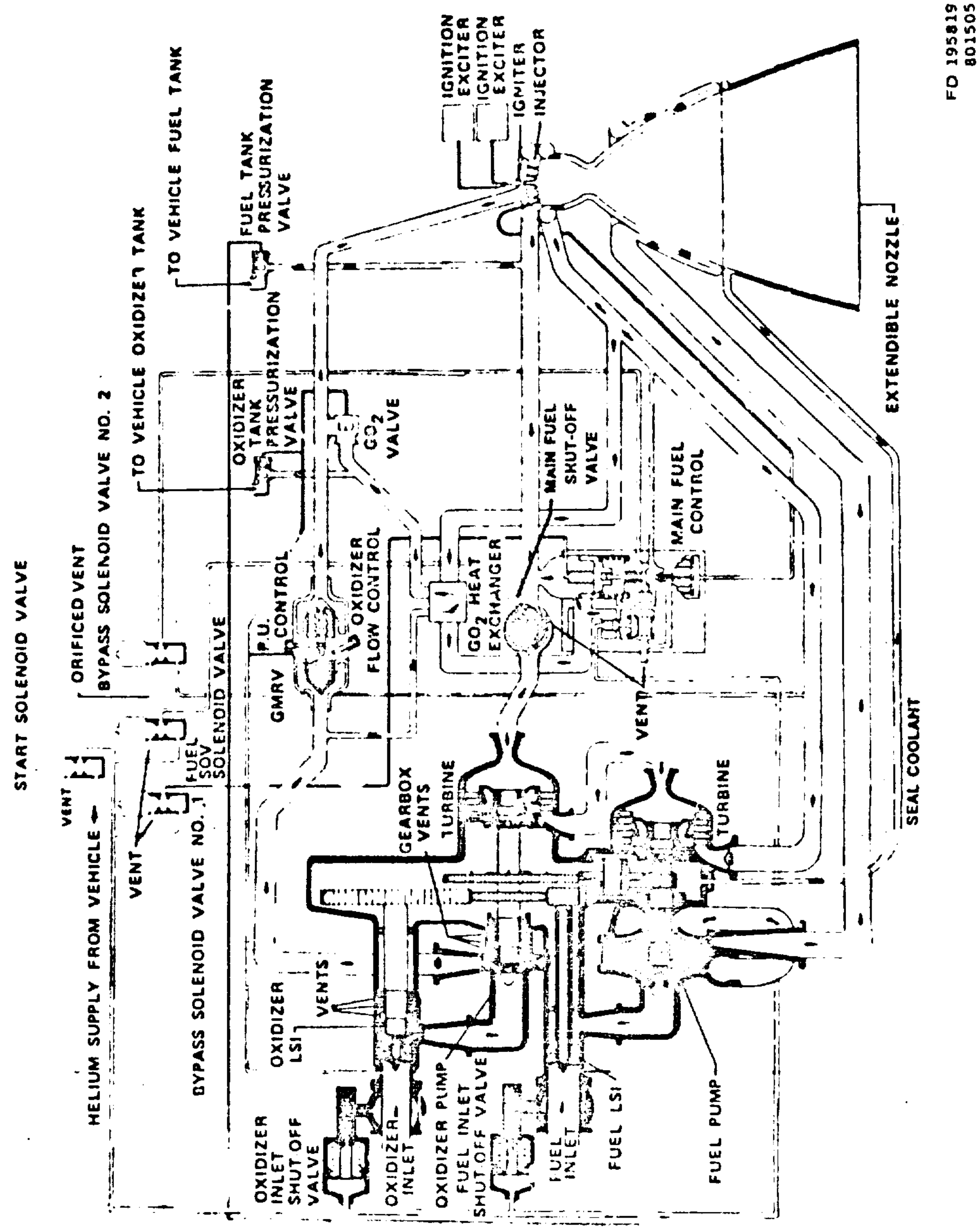

Figure 4-11. RL10 Category IV Engine Propellant Flow Schematic

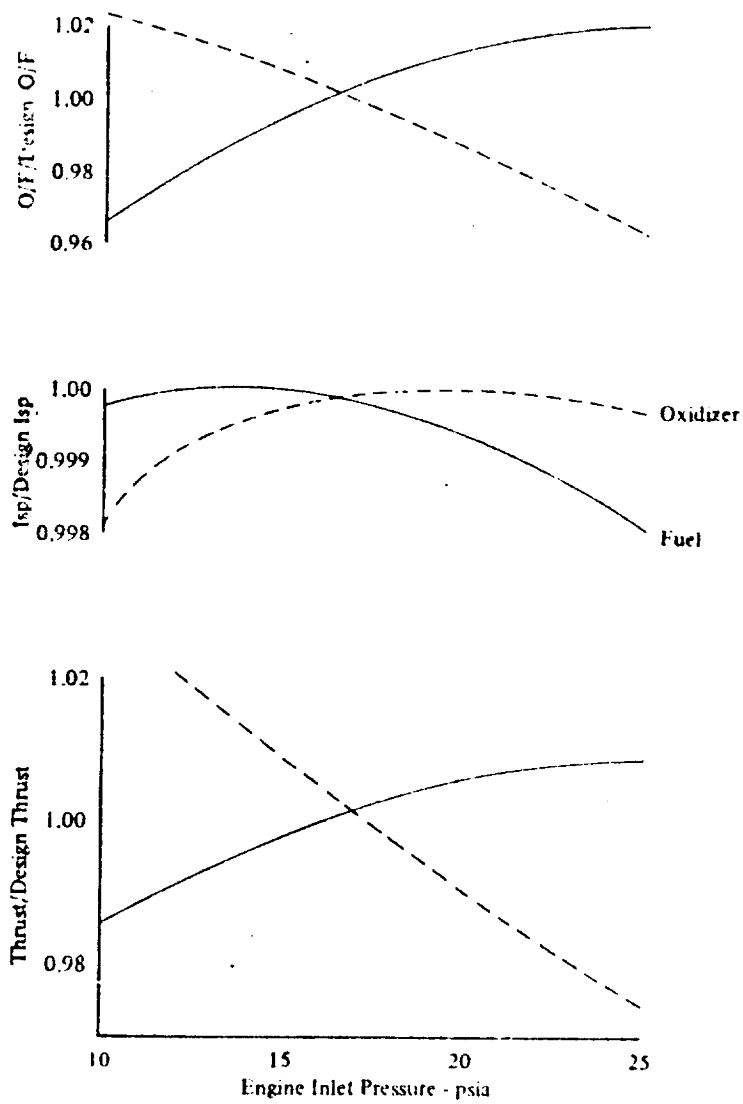

Figure 4-15. Inlet Pressure Effects on Advanced Expander Engine

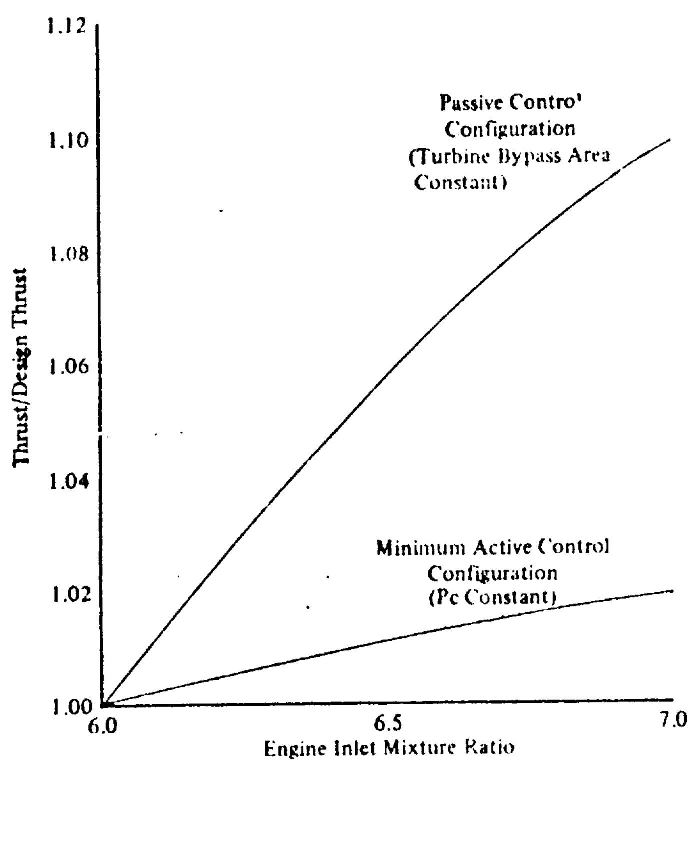

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Figure 4-16. Off-Design Mixture Ratio/Thrust Characteristics for Active and Passive Control Configurations

A minimum active control configuration would add approximately 2 lb in weight and 3 to 5 million dollars in engine development cost, and provide a somewhat lower reliability than a passive control configuration. Because the difference in performance between the two configurations is negligible for an OTV without a propellant utilization system, and the passive configuration has an advantage in cost and reliability, a passive control configuration was selected for the preliminary engine optimization.

4.3.3 Thermal Conditioning/Start Losses

Prior to starting an engine using cryogenic propellants, the turbopumps must be cooled to near liquid conditions. This can be accomplished using a tank head idle (THI) mode of operation. This is a pressure fed mode of operation with nonrotating turbopumps that can be used to settle propellants and provide low thrust levels for small ΔV changes in addition to conditioning the pumps. Since the THI mode can be used for propellant settling, conditioning losses only occur during the time required to condition the pumps after the propellants are settled. Conditioning time is a function of the initial temperature of the turbopumps and propellant flowrates during conditioning and should be on the order of 1 to 2 min for the Advanced Expander Engine. Specific impulse levels in THI are greater than 400 sec and are a strong function of the chamber pressure level (propellant flowrates). This relatively high impulse level during thermal conditioning minimizes losses, since the propellant losses are equivalent to:

$$\left(1.0 - \frac{Isp_{conditioning}}{Isp_{rated\ thrust}}\right) \times \int_{t_1}^{t_2} w_p$$

where:

t_1 = time propellants are settled
t_2 = time of completed conditioning
w_p = propellant flowrate

Propellant flowrates are related to required conditioning time and specific impulse in such a way to minimize losses whether flowrates are high or low. When propellant flowrates are high, specific impulse is high and conditioning times are relatively short. When flowrates are low, impulse is lower and conditioning times are relatively longer. However with lower flow rates, the total propellant quantity used is relatively low as the cooldown becomes more efficient. Acceleration transient losses are minimal. Impulse levels are between the THI and rated thrust levels and start transient durations should be less than 3 sec.

4.3.4 Preliminary Design Points

The results of the cycle optimization studies were incorporated in the engine design point definition computer program. Engine operating points were then defined for the nominal rated thrust condition of 10-, 15-, and 20-K advanced expander engines. A summary of these design points is presented in Table 4-2.

TABLE 4-2. PRELIMINARY ADVANCED EXPANDER ENGINE DESIGN POINT CHARACTERISTICS

Parameter			
Thrust, lb	10,000	15,000	20,000
Specific Impulse, sec	483.5	482.0	479.6
Inlet Fuel Flow, lb/sec	2.97	4.46	5.97
Inlet Oxidizer Flow, lb/sec	17.81	26.76	35.83
Chamber Pressure, psia	1,345	1,505	1,580
Mixture Ratio, Vehicle	6.0	6.0	6.0
Oxidizer Low Pressure Pump			
Inlet Total Pressure, psia	17.0	17.0	17.0
Inlet NPSH, ft	2	2	2
Discharge Pressure, psia	71	77	80
Speed, rpm	3,650	2,980	2,580
Efficiency, %	64	66	68
Specific Speed	1,150	1,060	1,020
Horsepower	5.6	9.1	12.4
Fuel Low Pressure Pump			
Inlet Total Pressure, psia	16.2	16.2	16.2
Inlet NPSH, ft	15	15	15
Discharge Pressure, psia	39	47	54
Speed, rpm	37,500	30,600	26,500
Efficiency, %	72	76	79
Specific Speed	4,565	3,645	3,115
Horsepower	5.7	10.9	17.3
Main Oxidizer Pump			
Inlet Total Pressure, psia	71	77	80
Inlet Temperature, °R	164	164	164
Discharge Pressure, psia	1,930	2,160	2,265
Speed, rpm	64,040	56,790	50,960
Efficiency, %	64.7	67.9	70.1
Specific Speed	1,420	1,420	1,415
Horsepower	192	306	415
Main Fuel Pump			
Inlet Total Pressure, psia	39	47	54
Inlet Temperature, °R	37.2	37.3	37.3
Discharge Pressure, psia	3,325	3,990	4,275
Speed, rpm	115,000	115,000	115,000
Efficiency, %	58.6	63.3	67.1
Specific Speed	565	605	670
Horsepower	1,045	1,685	2,225
Fuel Turbine			
Inlet Total Pressure, psia	2,865	3,440	3,685
Inlet Total Temperature, °R	967	884	818
Flow, lb/sec	2.78	4.23	5.69
Efficiency, %	60	62	64
Horsepower	1,020	1,660	2,200
Percent Admission	39.7	45.0	55.1
Pressure Ratio (total/static)	1.59	1.69	1.72
Oxidizer Turbine			
Inlet Total Pressure, psia	1,715	1,930	2,040
Inlet Total Temperature, °R	901	815	751
Flow, lb/sec	2.78	4.23	5.69
Efficiency, %	74	76	77
Horsepower	230	350	465
Percent Admission	87	94	100
Pressure Ratio (total/static)	1.09	1.10	1.10

TABLE 4-2. PRELIMINARY ADVANCED EXPANDER ENGINE DESIGN POINT CHARACTERISTICS (CONTINUED)

Thrust Chamber			
Chamber Pressure, psia	1,345	1,505	1,580
Mixture Ratio, chamber	6.20	6.14	6.10
Fuel Injector ΔP, psid	125	140	150
Oxidizer Injector ΔP, psid	200	225	225
Chamber Throat Diameter, in.	2.18	2.53	2.85
Primary Nozzle Expension Ratio	312	248	208
Nozzle Expansion Ratio	778	642	[illegible]
Nozzle Exit Diameter (ID), in.	60.8	64.1	66.5
Regenerator			
Effectiveness, %	36	43	48.5
Cold Side Pressure Loss, psid	5	5	5
Hot Side Pressure Loss, psid	7.3	8.1	8.6
Cold Side Temperature Rise, °R	260	275	285
Hot Side Temperature Drop, °R	290	305	310
Chamber/Nozzle Coolant			
Pressure Loss, psid	395	475	510
Temperature Rise, °R	610	510	445

SECTION 5

LOW-THRUST OPERATION

5.0 GENERAL

RL10 derivative and advanced expander cycle engine characteristics at low thrust were examined to determine the effects of extended low-thrust operation. The impacts on critical components and engine life were defined, and performance characteristics were generated. No modifications to the engines were required to enable extended operation at low thrust levels.

Kitting of critical engine components for the advanced expander cycle engine was investigated to define possible improvements in engine performance, life, and reliability for low-thrust operation. It appears that performance, weight, and/or reliability gains are achievable with a kitted version of the baseline advanced expander cycle engine specifically configured for low-thrust missions. The available gain and cost of kitting are provided.

5.1 RL10 DERIVATIVE ENGINES

The low thrust operational characteristics of the RL10 derivative engines were reviewed to determine the impact of extended operation at low thrust levels (750 to 3750 lb_f). "Extended operation" was defined as the time required to expend the propellants of a fully tanked orbital transfer vehicle (OTV) stage at the low thrust propellant flowrates. Therefore, stage burn time ratio is approximately:

$$\frac{\text{Low Thrust Burn Time}}{\text{Rated Thrust Burn Time}} = \frac{\text{Rated Thrust Level}}{\text{Low Thrust Level}} \times \frac{\text{Low Thrust Specific Impulse}}{\text{Rated Thrust Specific Impulse}}$$

For example, with an engine operating at the 10% thrust level, the engine firing duration will be approximately nine times as long as an equivalent-rated thrust firing duration.

5.1.1 Engine Life Capabilities

No modifications to the baseline designs of the derivative engines are required to enable extended operation at low thrust levels. Derivative engine accumulated firing duration beyond the specified life, is limited by turbopump gear and seal wear. The reduced stress requirements for these elements at low thrust more than compensate for the increase in required duration, resulting in more missions between engine overhauls than on high thrust missions. Engine low-cycle fatigue (LCF) life is also increased at low thrust levels due to the greatly reduced pressure strains (and, therefore, total strain) in thrust chamber passage walls. Since the number of engine thermal cycles per mission is expected to be independent of thrust level, the number of missions between engine overhauls would be greater at reduced thrust than at rated thrust.

5.1.2 Performance and Stability Characteristics

Only relatively minor modifications are required to enable the RL10 Derivative engines to operate at reduced thrust levels. These modifications are presented in Table 5-1 as a function of engine thrust level. A cavitating venturi is required between the fuel pump and the nozzle coolant inlet if the coolant inlet pressure is less than 188 psia (hydrogen critical pressure) to isolate boiling instabilities in the coolant passages from the fuel pump. A high-loss oxidizer injector (Δp four times baseline engine Δp) injector is required for the Derivative II engines below 25% thrust (3750 lb) to prevent combustion instability. Since the Category IV engine has a high-loss oxidizer injector in its baseline design, no modification of this engine is necessary to prevent combustion instability. Low thrust performance characteristics of RL10 derivative engines are shown in Figure 5-1. (This figure supercedes similiar information presented in the OTV Parametric Data Book, P&WA Report FR-12253 dated 1 October 1979.)

Note in Table 5-1 and Figure 5-1 that, in the range from 8 to 12% of design thrust for the Derivative IIA and IIB engines, there is an option of using gas-gas injection or a high loss injector. Based upon both energy release efficiency and feed system stability, gas-gas injection appears to be more desirable. The specific impulse advantage shown for gas-gas injection in Figure 5-1 represents the energy release efficiency advantage, which is approximately 0.7% at 10% thrust. The greater stability margin predicted for gas-gas injection is illustrated in Figure 5-2. Although both configurations are stable in the thrust range, the high loss injector is very near the unstable region.

TABLE 5-1. MODIFICATIONS TO RL10 DERIVATIVE ENGINES FOR LOW-THRUST OPERATION

Thrust Level (lb.)		*Weight Change (lb)*	*Required Modifications*
Derivative IIA, IIB	*Category IV*		
750-1200	750-1500	+4	Cavitating venturi added to isolate the fuel pump from jacket boiling instabilities. All oxidizer vaporized in gaseous oxygen heat exchanger providing gas-gas injection to prevent combustion instability and provide increased performance over liquid gas injection
1200-3750	N/A	+4	Cavitating venturi and high loss injector added. High loss injector required to prevent combustion instability
1200-1800	N/A	+15	If higher performance is desired in 8-12% thrust range (see Figure 5-1), a dedicated low thrust engine can be kitted by adding a special gaseous oxygen heat exchanger to vaporize all the oxidizer
3750-15000	1500-15000	0	None

5.2 ADVANCED EXPANDER CYCLE ENGINE

The 15K thrust point design engine defined during the engine optimization task of this study was selected as the baseline engine. A 10% (1500 lb) thrust level was selected as the low-thrust operating point, which was midway in the low thrust range of interest (1000 to 2000 lb thrust)./

5.2.1 Baseline Engine

Initial evaluation of low-thrust operation indicated that, with the regenerator location initially used, chamber coolant exit temperatures were unacceptably high (>1500°R). The temperature problem was resolved by moving the regenerator hot side from the injector inlet to the turbine discharge, as shown in Figure 5-3. This decreases the fluid temperature to acceptable levels because of the high turbine bypass flow level at low thrust (approximately 50% of available flow), which reduces the heat transfer capability of the regenerator. This configuration change has no significant effect on full-thrust operation because of the low turbine bypass flow level (97% of the hydrogen still passes through the regenerator hot side). Turbine inlet temperature characteristics at 10% thrust are shown in Figure 5-4 as a function of inlet mixture ratio.

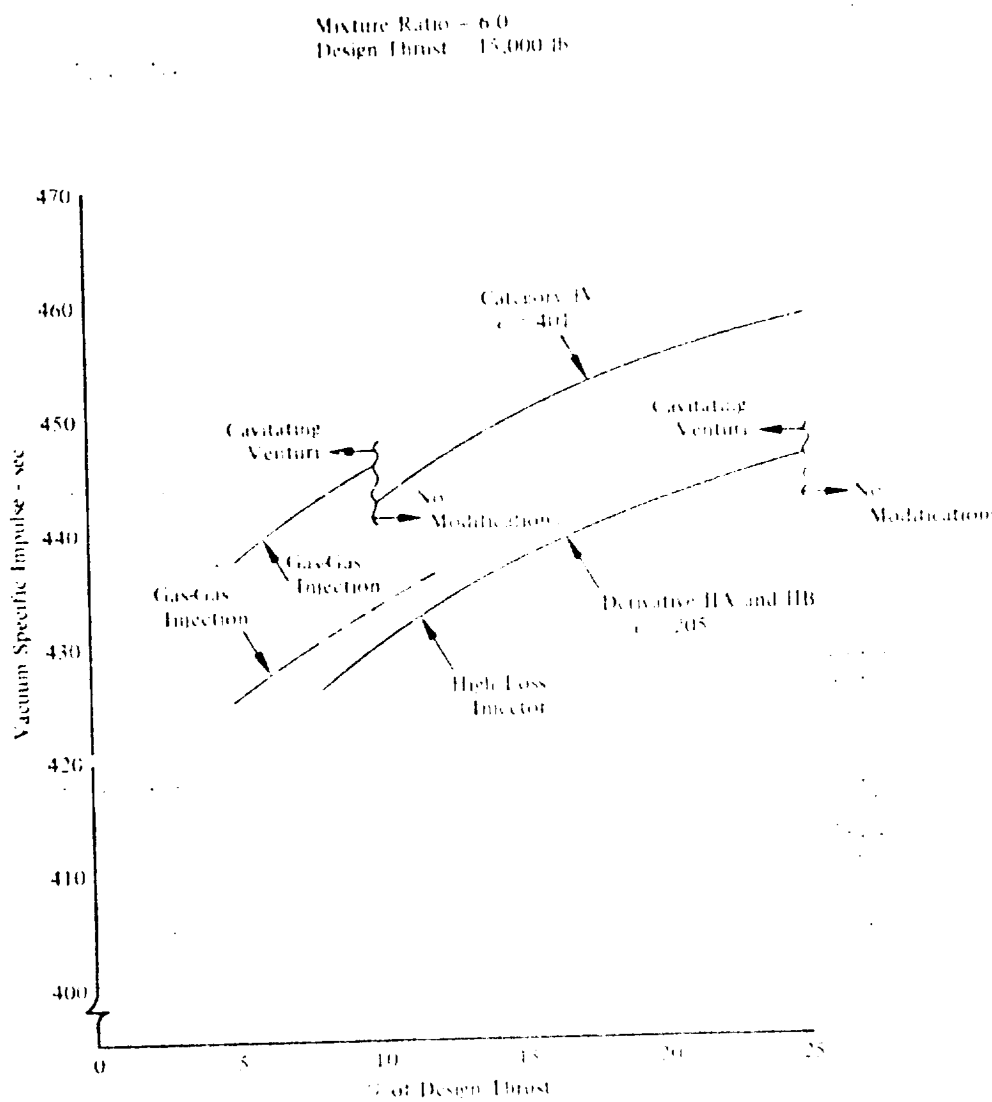

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Figure 5-1. RL10 Derivative Engine Low Thrust Performance Characteristics

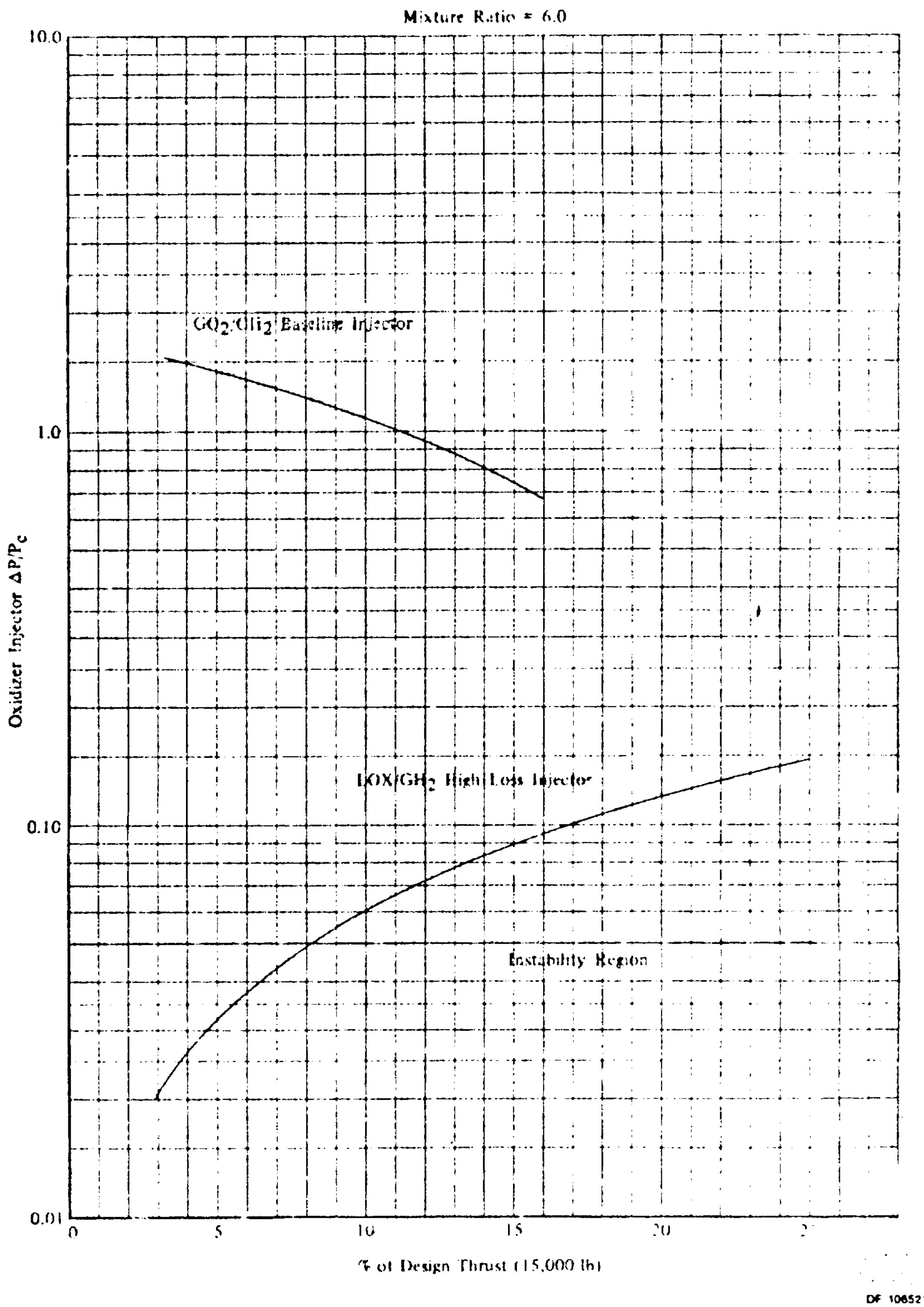

Figure 5-2. RL10 Derivative II Engines Injector Stability Characteristics

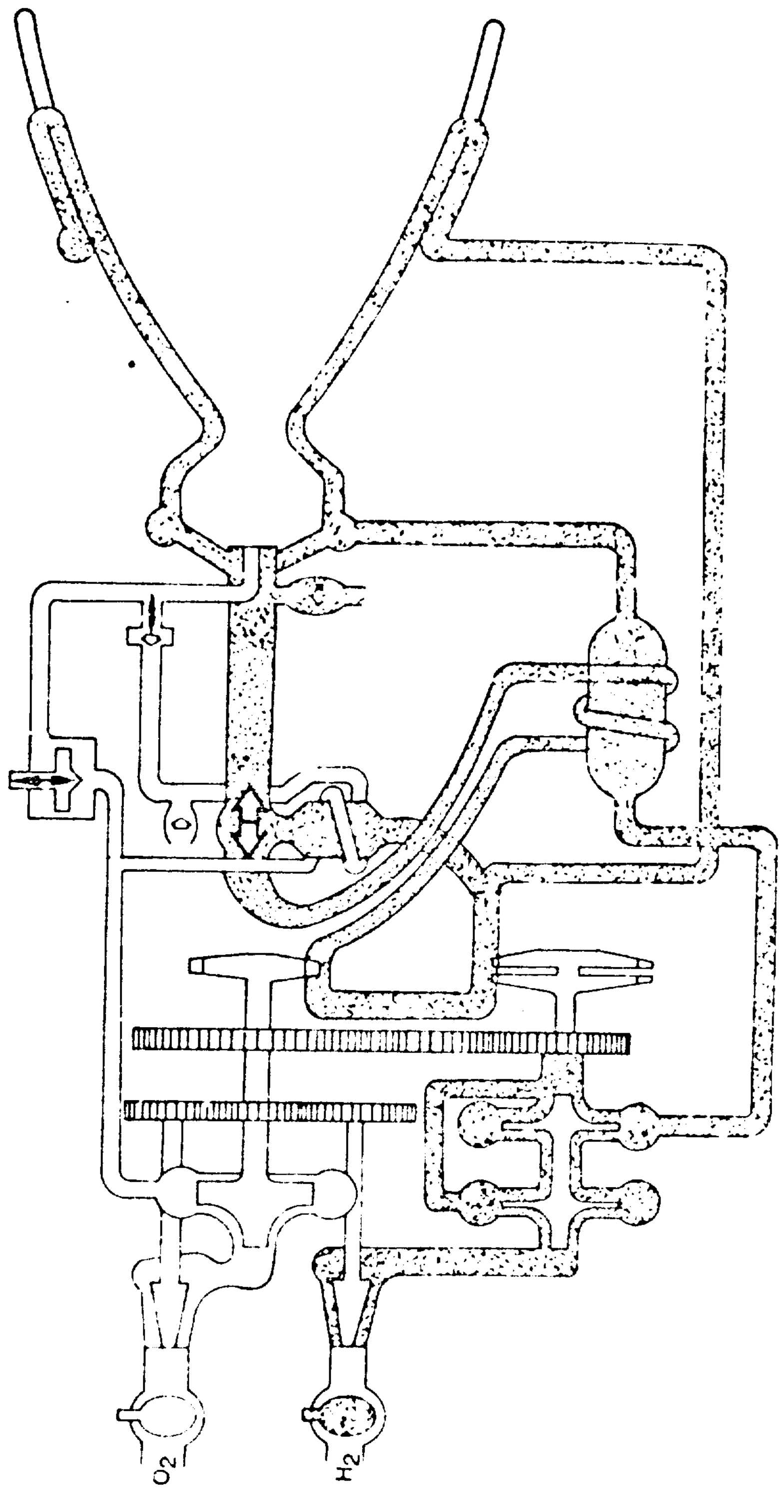

Figure 5-3. Revised Baseline Advanced Expander Engine

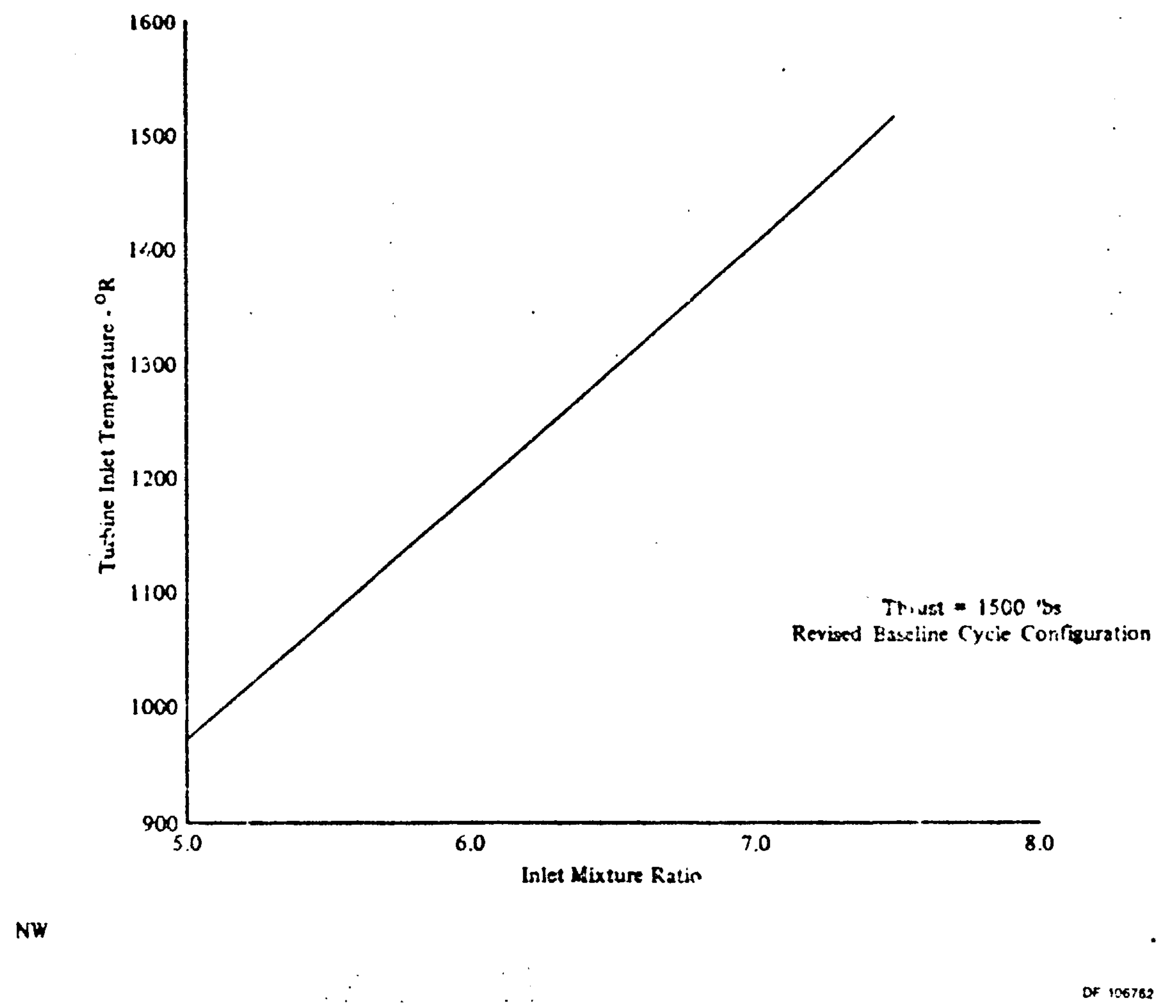

Figure 5-4. Advanced Expander Engine Mixture Ratio Effects on Turbine Inlet Temperature at Low Thrust

5.2.1.1 Life

Engine life characteristics were evaluated, and no additional modifications are required to enable extended operation at low thrust levels. Engine-accumulated firing duration is limited by turbopump gear and seal wear. The reduced stresses on these elements at low thrust more than compensate for the increase in required duration, resulting in more missions between overhauls than on high-thrust missions. Engine low-cycle fatigue (LCF) life is increased significantly at low thrust levels. Thermal strains are reduced because of lower hot-wall temperatures (approximately 250°R at the same O/F), and the pressure strains are decreased to an insignificant level. Since the number of engine thermal cycles per mission are expected to be relatively independent of thrust level, the number of missions between engine overhauls would be greater at reduced thrust than at rated thrust.

5.2.1.2 Stability

Fuel system boiling instabilities and low-frequency combustion instability are sometimes a concern in low-thrust off-design engine operation. Fuel system boiling instabilities (pressure oscillations) occur when rapid liquid-to gas-phase change is initiated due to heat addition at pressures less than critical pressure (188 psia for hydrogen). These pressure oscillations, if not isolated from the turbopump by some means (usually a cavitating venturi), can cause fuel pump stall. Combustion instability (chugging) results when injector pressure loss is too low ($\Delta P/P_c < 0.05$) to isolate the injector from the combustion process. With a liquid injector $\Delta P/P_c$ is reduced proportionally as thrust is decreased, requiring a 0.50 $\Delta P/P_c$ injector pressure loss at full thrust to provide an acceptable injection pressure loss ($> 0.05\ \Delta P/P_c$) at 10% thrust.

Oxidizer injector pressure loss is maintained at greater than 0.05 $\Delta P/P_c$ at low thrust by utilizing the GOX heat exchanger to vaporize the oxidizer before injection. This not only provides adequate injector pressure loss for stability, but also increases specific impulse at the 10% thrust level by approximately 7 sec, due to increased combustion efficiency. Figure 5-5 shows injector $\Delta P/P_c$ as a function of inlet mixture ratio for both propellant injectors.

5.2.1.3 Controls

A passive control configuration was selected for low-thrust operation. During the engine optimization task, passive and minimum active control configurations were evaluated for full thrust operation, and the passive (open-loop) control configuration was selected on the basis of simplicity. Control system simplicity is a primary concern in a new engine design, because closed-loop controls are more expensive, less reliable, and require substantially more testing than open-loop controls.

Transients are controlled with an open-loop system by ramping control valves with time or engine pressure levels. This type of transient control has been used successfully by the RL10 engine. The expander cycle engine with the turbopumps geared together lends itself to this kind of control because of its inherent power limit.

For an open-loop control system, control valve areas are set during acceptance testing by trimming the engine to the desired thrust and mixture ratio levels at nominal vehicle propellant inlet conditions. During flight operation, propellant inlet conditions can vary from nominal levels resulting in mixture ratio, thrust, and specific impulse variations. Figure 5-6 shows the effects of inlet pressure variations on engine performance. Less than 5% variation in thrust and mixture ratio and 0.5% in specific impulse result from the extremes of expected propellant inlet pressure variations.

1500 lb thrust

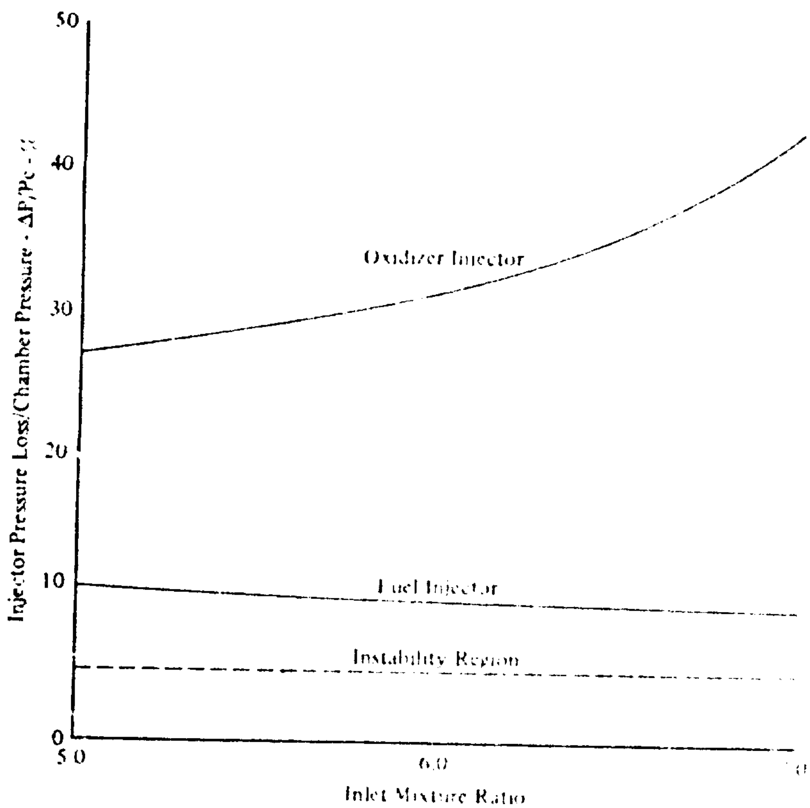

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Figure 5-5. Injector Stability Characteristics During Low-Thrust Operation, Advanced Expander Engine

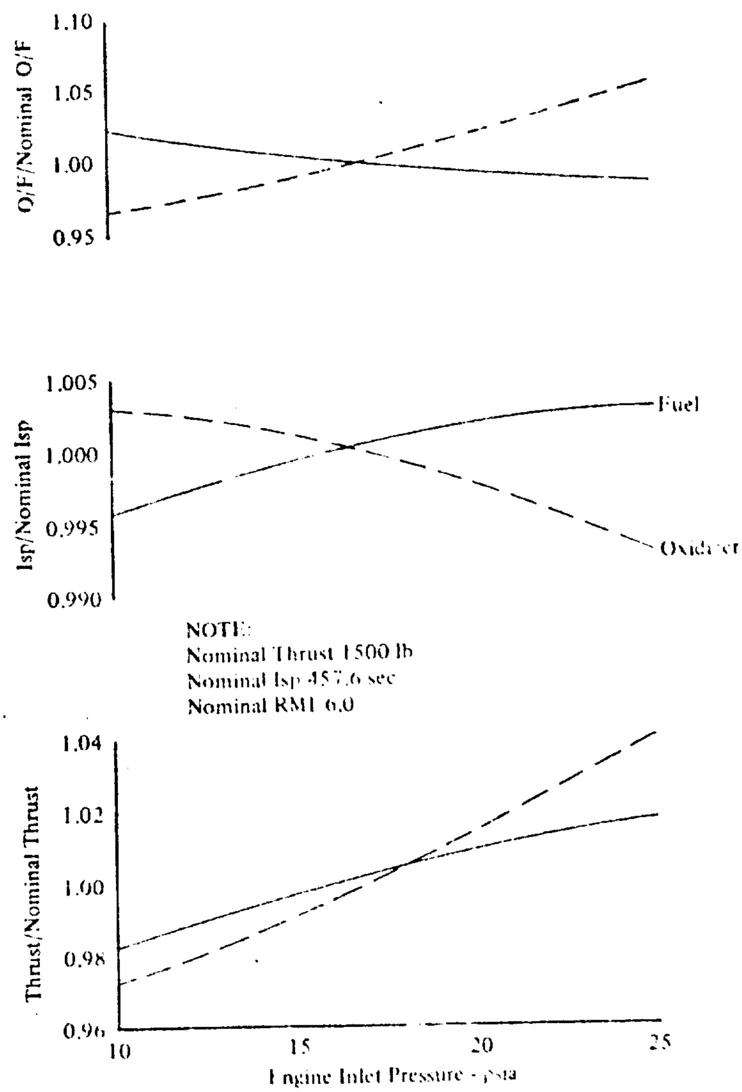

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Figure 5-6. Inlet Pressure Effects on Advanced Expander Engine Performance During Low-Thrust Operation

5.2.1.4 Performance

Low-thrust specific impulse characteristics for the baseline engine are presented in Figure 5-7 as a function of thrust and mixture ratio. The 2 to 5% drop in specific impulse levels between full thrust and low thrust is almost entirely due to increased kinetic losses. Boundary layer and divergence losses remain essentially constant on a delta impulse basis. Combustion losses decrease due to the increased combustion efficiency resulting from the gas-gas injection, and ideal impulse increases because of the increased injector enthalpy due to greater relative heat pickup (enthalpy pumping) in the chamber/nozzle coolant.

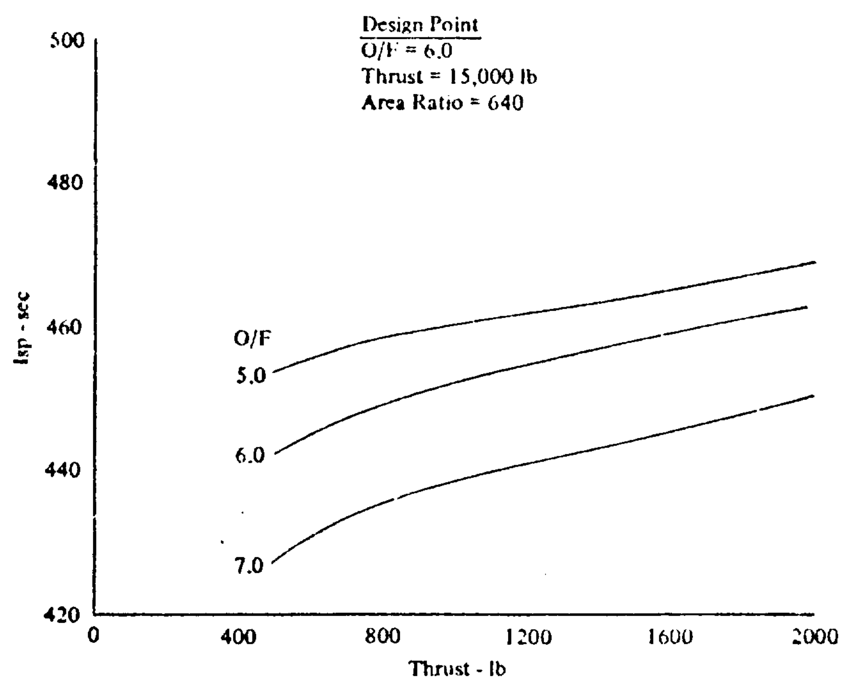

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Figure 5-7. Advanced Expander Engine Off Design Low Thrust Characteristic, Revised Baseline Cycle Configuration

5.2.2. Kitted Engine

Kitting of critical engine components for low-thrust operation was investigated to define possible improvements in engine performance, life, and reliability. Baseline engine life at low-thrust operation already significantly exceeds all known requirements due to the less severe operating conditions, making kitting for life purposes not practical. Kitting of the turbomachinery was not considered because of the major cost impact. The results of the kitting investigation, including cost where applicable, are presented in Table 5-2 and discussed in the following paragraphs.

TABLE 5-2. KITTED BASELINE ENGINE SUMMARY

Component Kitted	Effect	Weight Change (lb)	Cost Impact DDT&E ($M)	Cost Impact Production ($M)
1. Controls	Increased Reliability	−16	+1.0	+0.1
2. Chamber/Nozzle	Optimized Design; +14.5 sec Isp	−35	+51	+0.6
3. Regenerator Redesign	Optimized Design	−18	+1.5	+0.1

Note: 1. Costs are rough-order-of-magnitude in FY'79 millions and are increases above baseline engine DDT&E levels not considering required consumables or facility modifications.

2. Production costs are per engine cost based on a buy of 50 kitted engines.

5.2.2.1 Controls

The controls could be kitted to increase the engine reliability potential and decrease engine weight relative to the baseline engine. The fuel control valve of the baseline engine is a three-position valve. If full-thrust operation is not required, the valve could be changed to a two-position valve configuration. This would eliminate the actuation system required for the third position, which would slightly reduce engine weight and increase the reliability by reducing the possible failure modes. An even more significant advantage can be gained on the oxidizer system. If full-thrust operation is not required, the oxidizer liquid flow system can be eliminated, as shown in Figure 5-8. This change would also increase reliability by further reducing the possible failure modes and decrease engine weight. A reduction in engine weight of approximately 16 lb can be realized by kitting the fuel control and removing the oxidizer liquid control system.

5.2.2.2 Chamber/Nozzle

Kitting of the chamber/nozzle would provide reduced weight and increased specific impulse. Because of the high power margin at low thrust, the chamber/nozzle could be changed to a counterflow configuration instead of the baseline engine parallel flow configuration (shown in Figure 5-9). This allows removal of the regenerator because with the counterflow configuration, vaporization of the hydrogen, before it is used to cool the chamber/nozzle, is no longer required. Chamber throat area can also be reduced and chamber pressure increased until the fuel pump stall line is reached or the engine runs out of power. Figure 5-10 shows the fuel pump characteristics at low thrust for inlet mixture ratios of 5.0, 6.0, and 7.0. For this design, the engine power limit was reached before reaching the pump stall limit due to the significant cycle power loss resulting from the regenerator removal. The increase in chamber pressure provides a slight decrease in weight and a significant increase in specific impulse due to the increased area ratio, increased injector enthalpy, and reduced kinetic losses. Figure 5-11 presents the specific impulse resulting from the higher chamber pressure levels as well as baseline engine performance. Kitting the chamber/nozzle provides a 35-lb reduction in weight and a 14.5-sec increase in specific impulse for the 1500-lb thrust mixture ratio of 6.0 condition. In conjunction with kitting the chamber/nozzle assembly, the injector could be kitted to provide a slight decrease (~5 lb) in engine weight. However, because of the high efficiency of gas-gas injection of the baseline engine at low thrust, no increase in engine *performance* would be realized from the injector change.

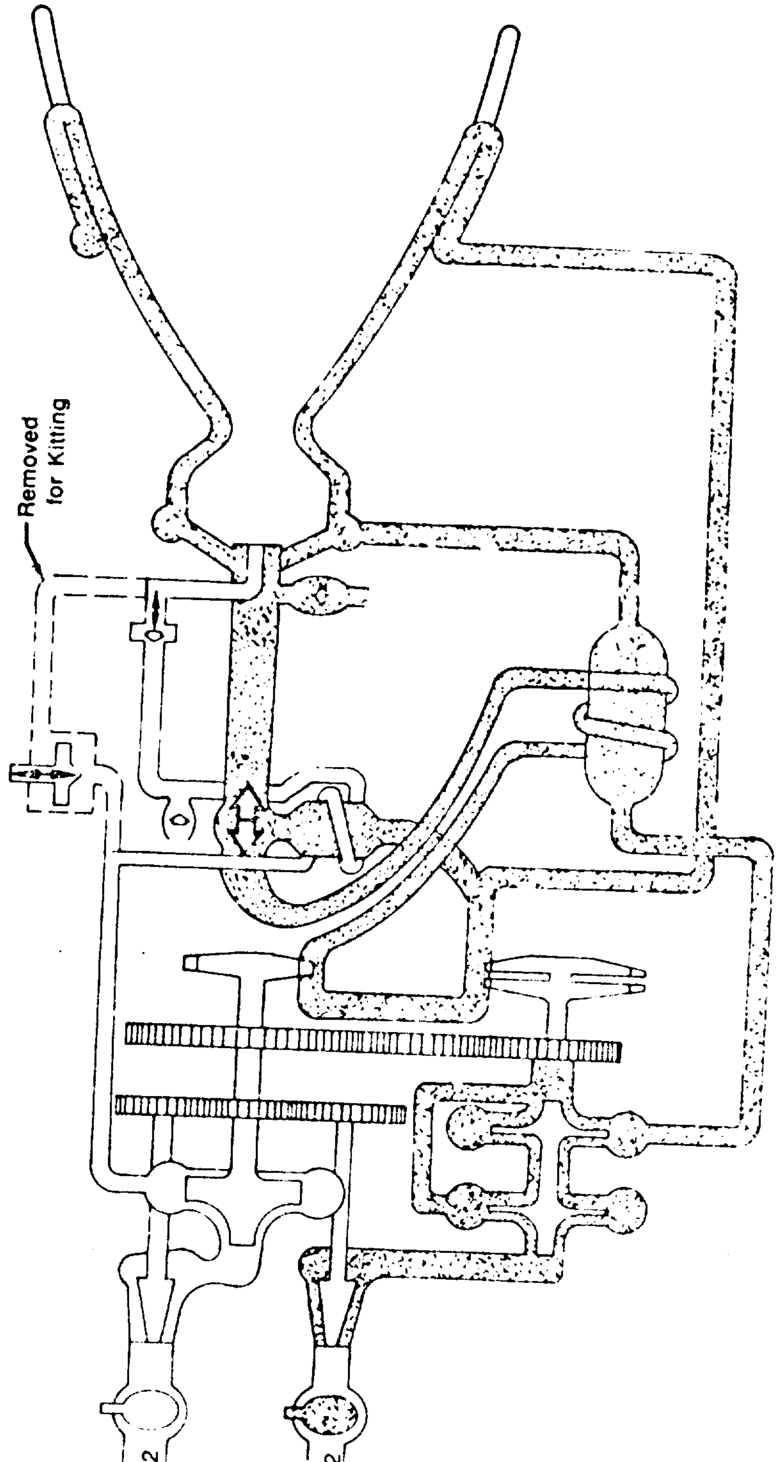

Figure 5-8. Advanced Expander Engine With Controls Kitted for Low-Thrust Operation

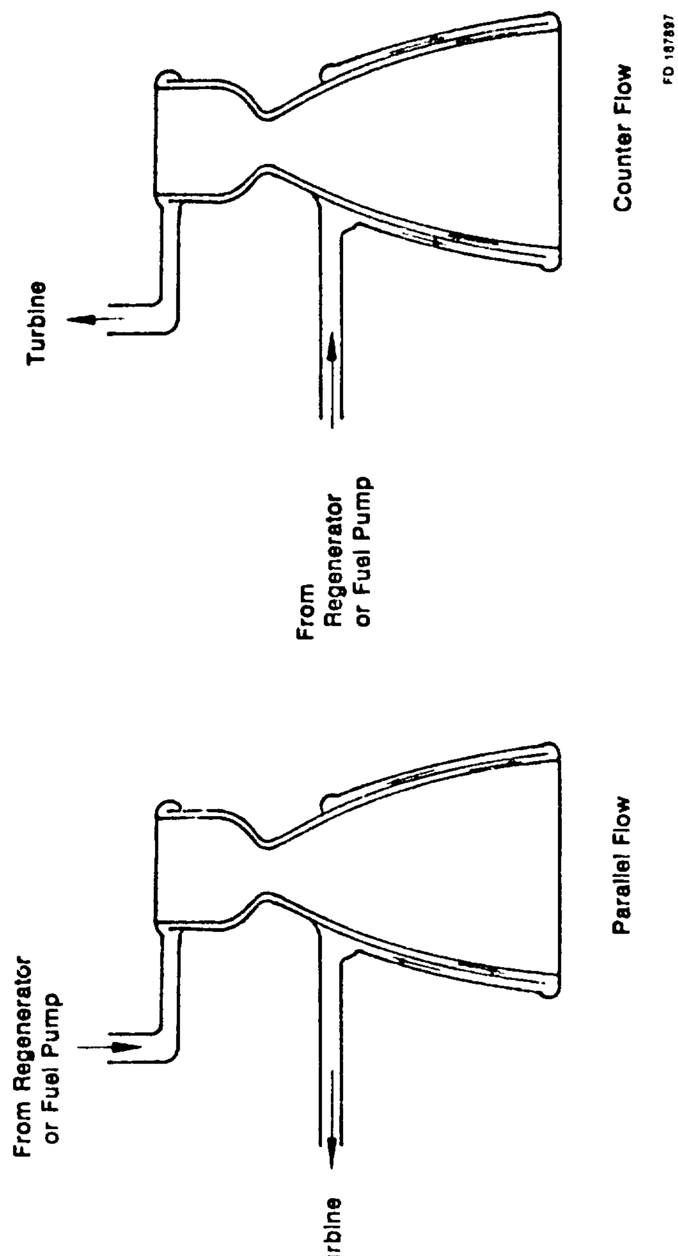

Figure 5-9. Thrust Chamber Regenerative Cooling Schemes

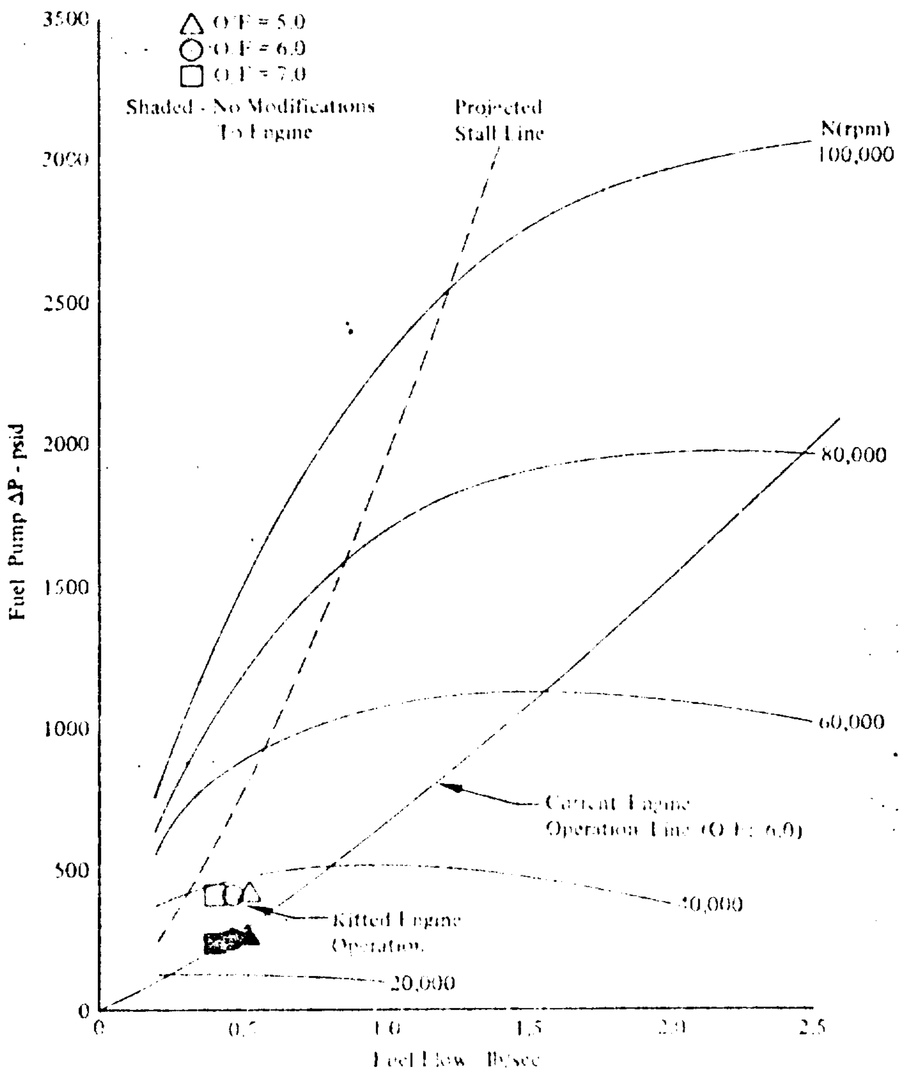

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Figure 5-10. Effect of Chamber/Nozzle Kitting on Fuel Pump Operation at Low Thrust, Advanced Expander Engine

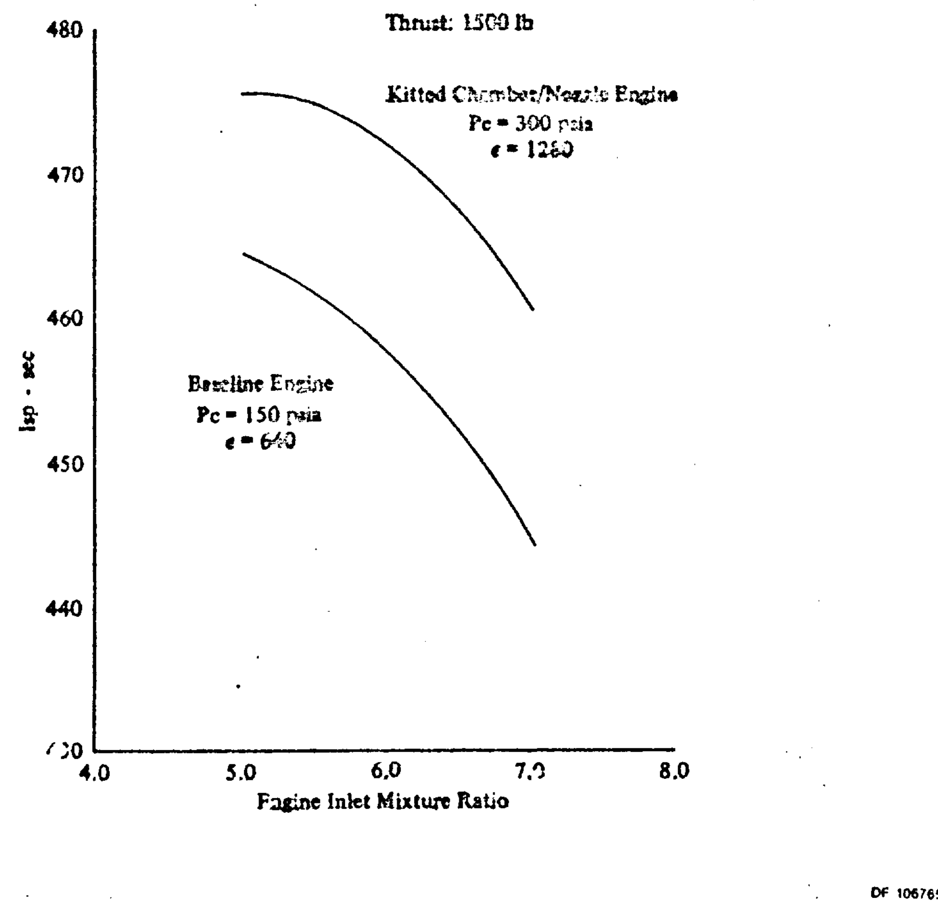

Figure 5-11. Low-Thrust Kitted Engine Performance Characteristics, Advanced Expander Engine

5.2.2.3 Regenerator

The regenerator could be kitted to decrease engine weight. Because of the high power margin experienced (only 50% of the available fuel flow is required to drive the turbine) during low-thrust operation, the regenerator designed for full thrust supplies a surplus amount of heat transfer. With the parallel flow chamber cooling configuration, the regenerator is required to prevent thrust chamber hot-wall burnout. Replacing it at low thrust with a smaller design, allows the regenerator to provide just enough heat transfer to vaporize the fuel prior to using it to cool the thrust chamber. This optimized low-thrust regenerator design requires 90% of the available fuel flow to drive the turbine and results in a decrease of 18 lb in engine weight.

SECTION 6.0

SAFETY AND RELIABILITY COMPARISONS

6.1 INTRODUCTION

Crew safety and mission reliability are important considerations in the selection of an engine configuration for the OTV. As part of the Task 6 effort, reliability comparisons were made for the 15K advanced expander cycle defined in this study and a staged combustion engine being considered for the OTV. Parametric curves of mission and crew safety reliability as a function of engine reliability were also generated for one, two and three engine OTV vehicles with, and without engine-out capability. Relative differences in crew safety and mission reliability were then estimated for the two engine cycles using the relative engine reliabilities and parametric curves established previously. The following sections describe the analyses made and results of the safety and reliability comparisons. Reliability definitions used in this study are given in Table 6-1.

TABLE 6-1. DEFINITIONS

1. Reliability
   - Projected — estimated lower limit based on sample size
   - Engine — probability of engine operating normally on next firing
   - System — probability of vehicle operating normally on next firing
   - Mission — probability of mission being completed as planned
   - Crew (safety) — probability of crew returning safely.
2. Confidence level — probability that true reliability is equal to or greater than estimated.
3. Accountable firings — engine tests that are valid for statistical reliability demonstration purposes and excludes tests outside expected flight engine conditions.

6.2 ENGINE SYSTEMS COMPARISON

Analysis was completed for vehicles employing one, two or three engines. A direct comparison of crew safety and mission reliability for these systems is shown in Figure 6-1. Assumptions for this analysis were:

1. Of all main engine failures, 10% will destroy adjacent engine(s) but not damage the remainder of the vehicle including the auxilliary propulsion system (APS)
2. Of all main engine failures, 5% will disable the vehicle
3. The auxiliary propulsion system (APS) has same reliability as main engine(s)
4. No rescue facilities are available
5. Six burns are required to complete mission
6. Three burns are required to return the crew
7. No vehicle damage results from APS failure.

Configuration	Mission Reqm't	Safety Reqm't	Mission Rel	Safety Rel
1 Main Engine	1 ME	1 ME	M1	S1
1 Main Engine, 1 APS	1 ME	1 ME or APS	M2	S2
2 Main Engines	2 ME	1 ME	M3	S3
2 Main Engines	1 ME	1 ME	M4	S4
2 Main Engines	2 ME	2 ME	M5	S5
2 Main Engines, 1 APS	2 ME	1 ME or APS	M6	S6
3 Main Engines	2 ME	1 ME	M7	S7
3 Main Engines	3 ME	2 ME	M8	S8

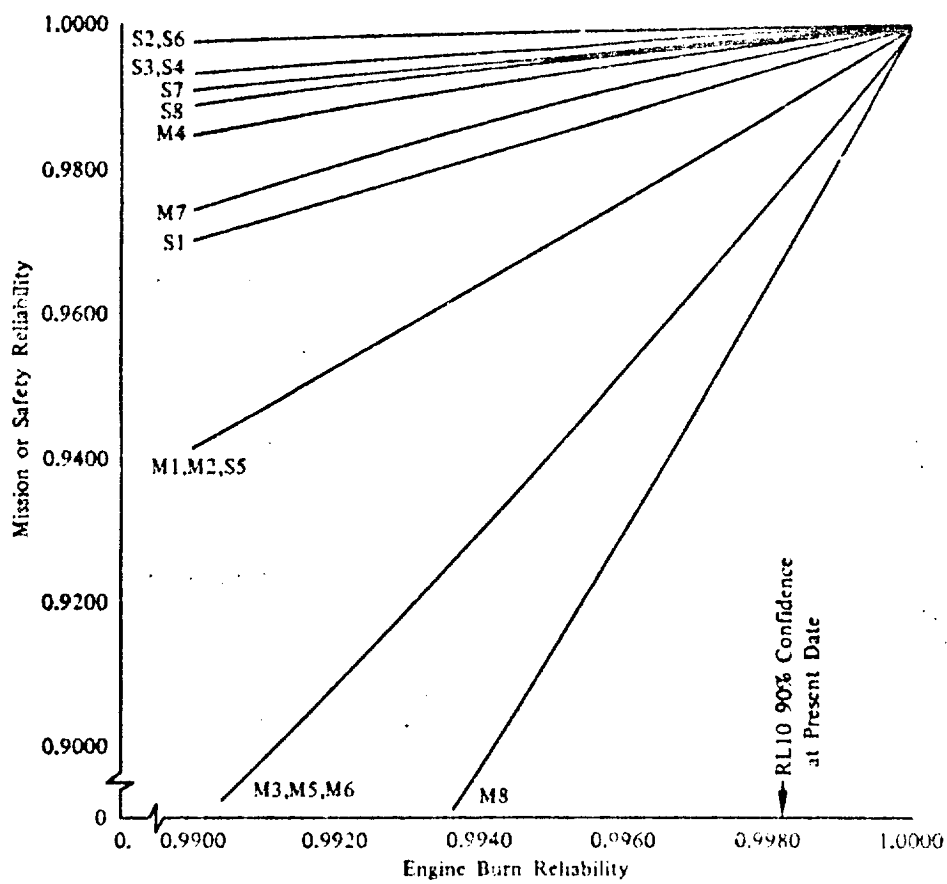

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Figure 6-1. Impact of Configuration on OTV system Reliability

Of the eight configurations presented in Figure 6-1, a two engine system with an engine-out safety capability and a single engine system with an APS safety backup provide the highest combined crew safety and mission reliability levels. Figure 6-1 also illustrates that engine reliability near the present RL10 demonstrated reliability of 0.9982 (90% lower bound confidence) is probably necessary to provide adequate mission and safety reliability.

Using assumptions similar to those proposed by Grumman as shown in Table 6-2, an engine system reliability of 0.9997 at the 90% confidence level is required for a man rated vehicle. Changes were made to the Grumman assumptions and the resulting reliability estimates are shown in Table 6-3.

1. Loss of entire OTV crew is 1 in 50 instead of single crew member
2. Losses due to engine are 15% instead of 50%
3. Losses in STS phase of mission are not considered in 1 in 50 crew career loss.

From Table 6-3, it is apparent that a multiple engine system with engine out capability requires a lower engine reliability. However, Figure 6-2 shows that to demonstrate this engine reliability would require more than 2200 accountable engine firings without a single failure.

6.3 ENGINE RELIABILITY

The inherent reliability of an engine is a function of its parts (e.g., controls, mechanical and structural design, cycle, etc.) and can only be demonstrated by firing the engine. Figure 6-3 shows demonstrated engine reliability at 90% confidence as a function of the number of firings. This figure shows that large numbers of engine firings are necessary to obtain high demonstrated engine reliability and any failure reduces demonstrated engine reliability significantly. As an example of the time required to obtain high reliability, demonstrated RL10A-3-3 engine reliability is shown on the figure.

Because of the excessive number of firings required to obtain the desired demonstrated engine reliability levels, the inherent reliability of the OTV engines should be maximized. This can be accomplished by reducing the number of failure modes (complexity) of the engines, eliminating single point failures wherever possible, and minimizing the number of catastrophe failure modes.

Control system simplicity is a primary driver in obtaining an engine with high inherent reliability. Figure 6-4 shows the relative control complexity as a function of the number of engine parameters on closed-loop control. This figure is conservative since only single measurement inputs were assumed but in typical application redundant measurements are utilized. Figure 6-5 (based on gas turbine engine electronic control reliability) relates the control complexity to the relative number of failures, indicating that the number of failures increases at faster than a one to one relationship with complexity. This intimates that unless the time and money are available to obtain a demonstrated reliability number by accumulating an excessive number of engine firings, a simple engine cycle with a minimum of closed-loop control functions should be selected for the OTV engine.

Single point failure modes can be reduced by providing redundant systems. However this is not always possible, and in those cases the system should be made fail safe, and additionally should be designed to operate even with a minor malfunction (e.g., strengthening the actuators on a valve so that it operates even if internal binding is present).

TABLE 6-2. ASSUMPTIONS FOR MOTV MAN RATING*

● MOTV crew member career risk (i.e., crew career survival rate = 0.98)	1 in 50
● Assumed number of missions per crew member	10
● Hence, crew mission survival rate.	0.998 (1 failure in 500)

This per mission survival figure has to be allocated between the STS and MOTV phases — making an arbitrary, even, division we have:

● Survival rate for STS phase of mission	0.999 (1 failure in 1000)
● Survival rate for MOTV phase of mission.	0.999 (1 failure in 1000)

Preliminary Allocation of MOTV Catastrophic Failure Likelihood

Critical Subsystem	*% Allocation*	*Missions per Catastrophic Failure*
Main Propulsion	50	2,000
RCS	10	10,000
EPS	8	12,000
Avionics	7	14,000
ECLS	10	10,000
Radiation Protection	12	8,000
Crew Transfer	3	30,000
Food/Water	0	∞
Overall Structure	0	∞

*Reference: *Manned Geosynchronous Mission Requirements and Systems Analysis Study*, Grumman Aerospace Corporation, 7 November 1979.

TABLE 6-3. REQUIRED OTV ENGINE RELIABILITY

Configuration	*Engine Reliability*	*System Reliability*
	1 Burn	3 Burns
Single Main Engine	0.9999	0.9997
Two Main Engines with Engine-Out Capability for Three Crew's Safe Return	0.9995	0.9997
One Main Engine with APS Backup for Crew's Safe Return	0.99899	0.9997

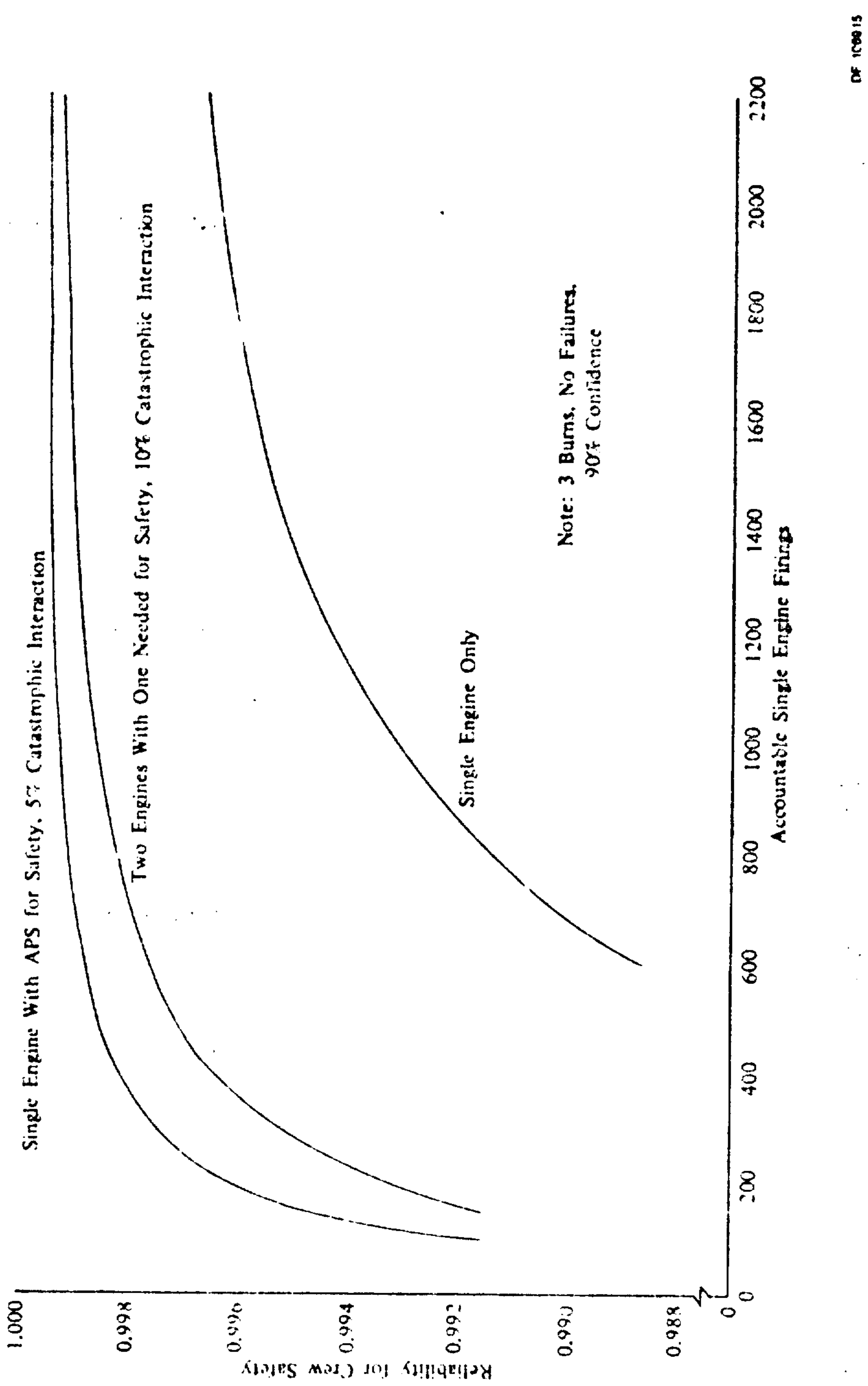
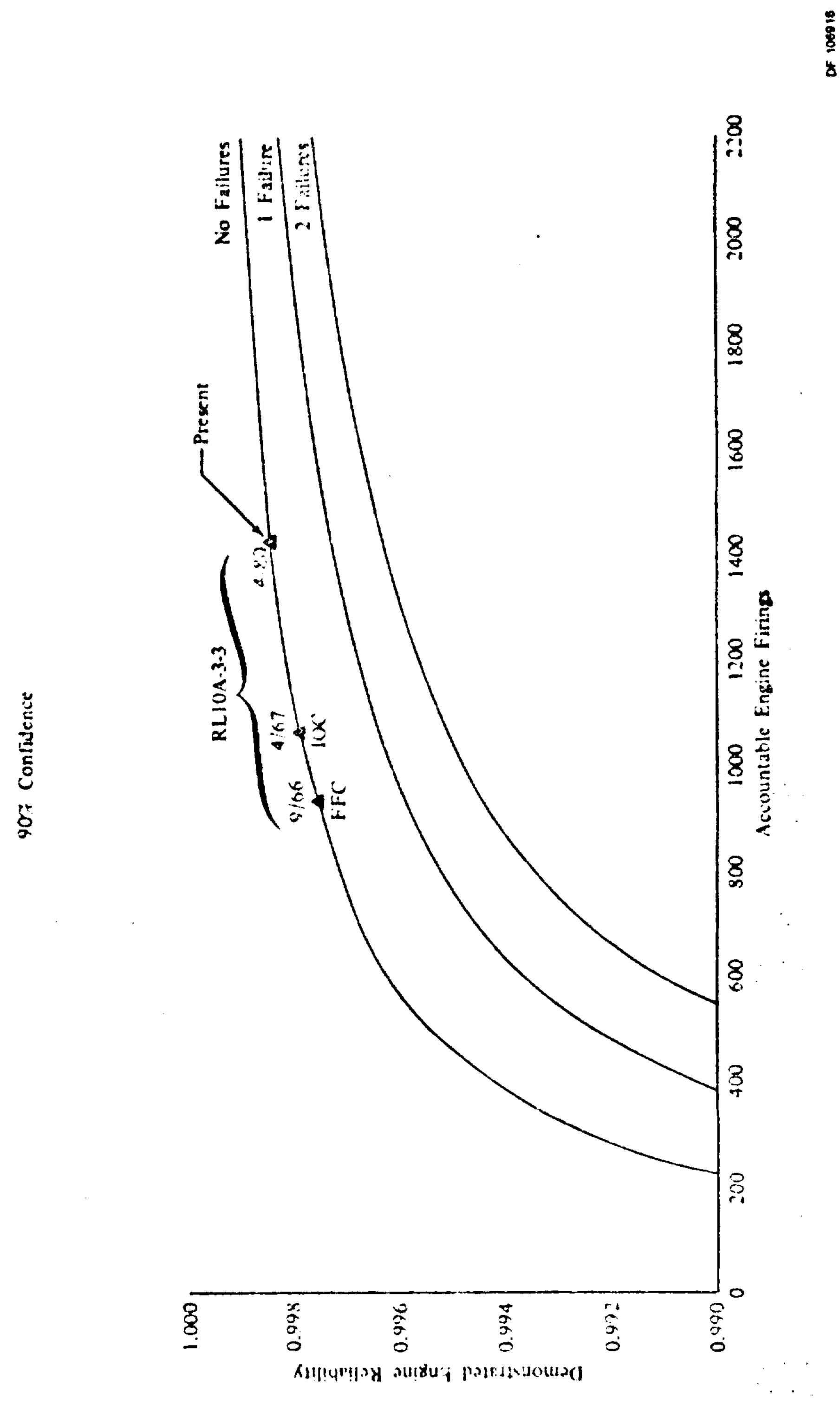


Figure 6-2. System Safety as a Function of Engine Firings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90% Confidence
No Failures
1 Failure
2 Failures
Present
RL10A-3-3
9/66
FFC
4/67
IOC
1.000
0.998
0.996
0.994
0.992
0.990
Demonstrated Engine Reliability
0
200
400
600
800
1000
1200
1400
1600
1800
2000
2200
Accountable Engine Firings

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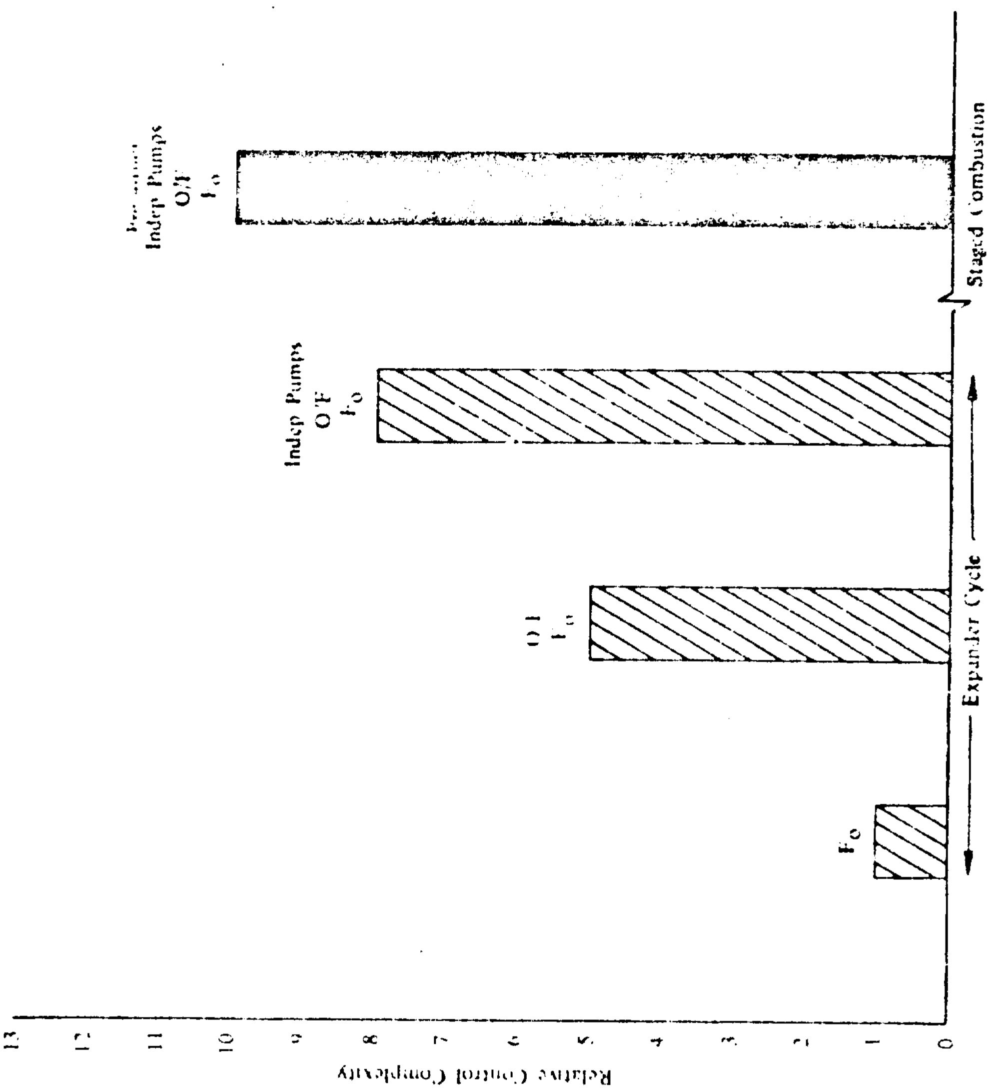


Figure 6-4. Steady State Control Function Effect on Complexity

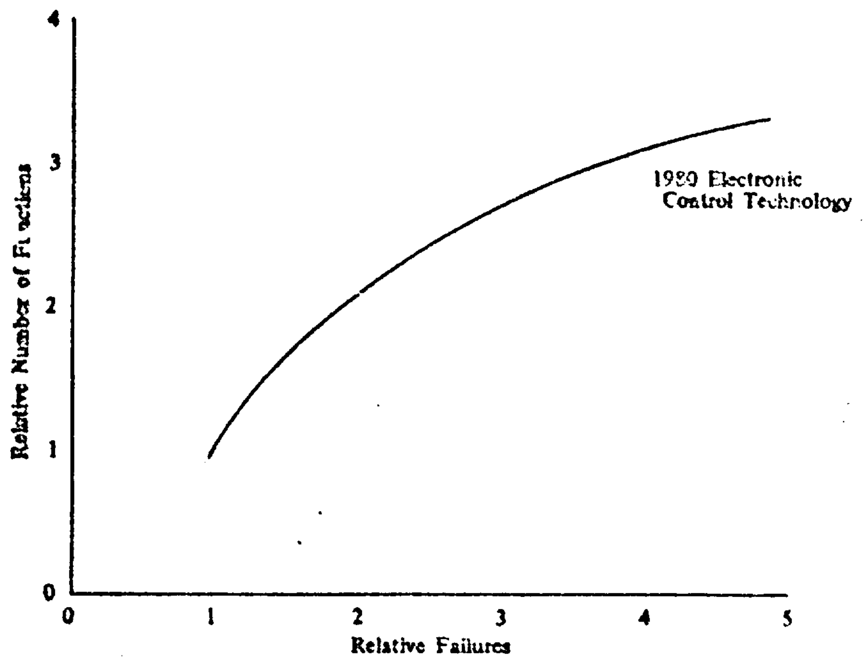


Figure 6-5. Electronic Control Reliability

The number of catastrophe failure modes of an engine is a function of the engine cycle and the control system. An expander cycle engine has fewer potential catastrophe failure modes than a staged combustion cycle engine because it is power limited and requires fewer closed-loop control functions. Gearing the turbopumps together also increases the inherent reliability of an engine because it reduces the number of control functions and also prevents one of the turbopumps from accelerating and causing a catastrophe failure. A series turbine engine configuration reduces the flow control requirements which increases the inherent reliability. The most inherently reliable engine configuration would be an expander cycle with open-loop control, a series turbine configuration, and the turbopumps geared together.

6.4 FAILURE MODE COMPARISON FOR ADVANCED EXPANDER CYCLE AND STAGED COMBUSTION ENGINES

The staged combustion engine evaluated in this task was a 20K thrust engine defined under Contract NAS8-32996. It was compared with the 15K thrust advanced expander cycle engine optimized in Task 4 of this study. Operating characteristics and the cycle for the staged combustion cycle engine are shown in Figures 6-6 and 6-7. Similar information for the expander cycle engine has been previously discussed.

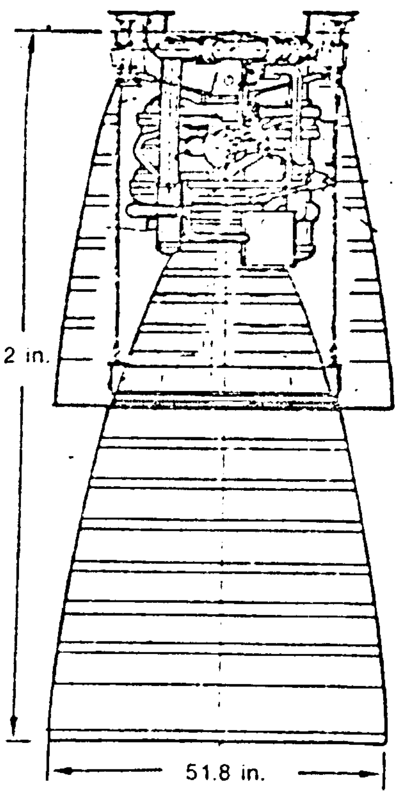
To accomplish the reliability comparison, cycle pressures and temperatures and engine design and operating characteristics were evaluated for the two configurations. A comparison of cycle parameter levels for the two configurations is shown in Table 6-4 while a comparison of design features for each is shown in Table 6-5.

The staged combustion engine operates at a chamber pressure of 2000 psia while the expander cycle operates at 1580 psia. Because of its higher chamber pressure, the staged combustion engine generally has higher system pressures and temperatures than the expander cycle. It utilizes combustion products burned at a low mixture ratio in a preburner to drive its main pumps while the expander cycle uses hydrogen heated in the regenerative chamber and nozzle and a regenerator to drive its pumps. The staged combustion engine utilizes an active closed-loop electronic control system while the expander cycle engine uses solenoid operated valves and hydromechanical controls with no closed-loop functions for its control. In the staged combustion engine all pumps operate independently while in the expander cycle engine all pumps are geared together eliminating the need for an active control system.

Design features and operating characteristics for the Advanced Expander engine were evaluated to identify its failure modes. A failure mode and effects analysis (FMEA) for the RL10A-3-3 engine modified to reflect differences for the Advanced Expander engine was used as the basis. A total of 66 failure modes were identified for this engine.

These failure modes were then evaluated as to potential hazard and classified as to the likelihood of occurrence. Definition of the classifications used to grade the failure modes by severity and likelihood of occurence are shown in Table 6-6 and 6-7. Previous RL10 experience with similar failure modes as well as advanced expander engine operating conditions were considered in classifying the failures. It was assumed that redundant vehicle propellant valves and helium shutoff valves would be available. This would eliminate the hazard of all engine failures related to loss of vehicle propellants or helium. The failure modes identified and hazard classifications assigned are shown in Table 6-8.

Only four failure modes for the Advanced Expander engine were found to be in the No. 1 or No. 2 hazard category (likely to cause complete system loss or major system damage). Based on this number of hazardous failure modes it was estimated that only 6% of the failures for an Advanced Expander engine would result in damage to an adjacent engine. It was further estimated that only 3% of the engine failures would damage the vehicle.



Thrust, lb	20,000
Chamber Pressure, psia	2,000
Expansion Area Ratio	165/400
Envelope	
Length, in.	60/112
Diameter, in.	49.9
Nozzle Percent Length	99.8
Specific Impulse, sec	
Mixture Ratio 6:1	477.1
Mixture Ratio 7:1	469.9
Engine Weight, lb	455
AMOTV Payload, lb	14,404
Full Thrust NPSH (O_2/H_2)	2/15
Tank Pressurization	Pumped Idle
Conditioning	Tank-Head Idle

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Figure 6-6. Staged Combustion Engine Evaluated in Task 6, Safety and Reliability Comparison (from Contract NAS8-32996)

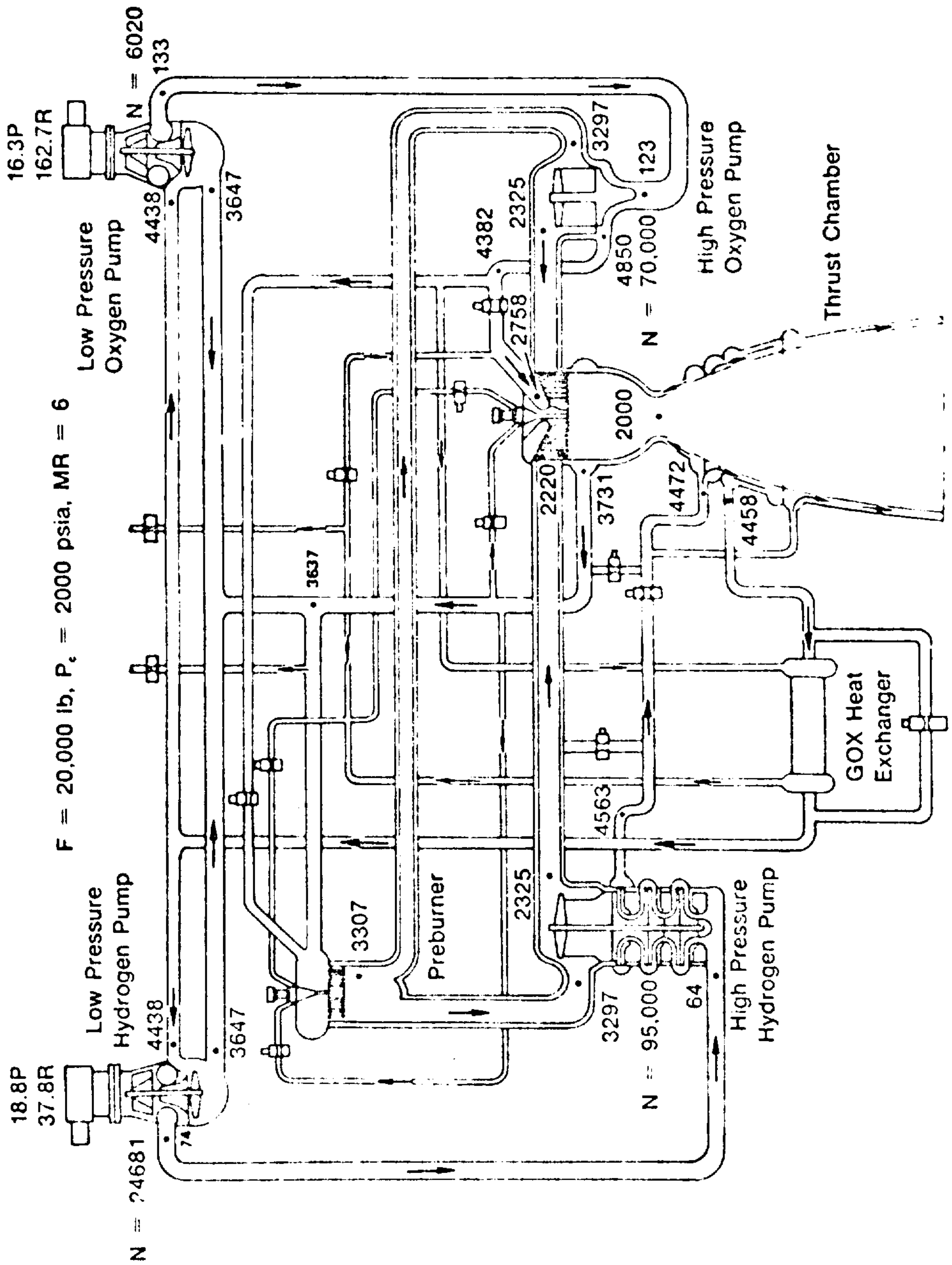

Figure 6-7. Staged Combustion Cycle Schematic Evaluated for Safety and Reliability Comparison

TABLE 6-4. COMPARISON OF ADVANCED-EXPANDER AND STAGED-COMBUSTION CYCLE PARAMETERS

	Advanced Expander	*Staged Combustion*
Thrust, lb	15,000	20,000
Chamber Pressure, psia	1505	2000
Chamber Mixture Ratio	6.14	6.43
Oxidizer Boost Pump Discharge Pressure, psia	77	133
Oxidizer Boost Pump Speed, rpm	2980	6020
Oxidizer Boost Pump Turbine Inlet Pressure, psia	—	4438
Oxidizer Boost Pump Turbine Inlet Temperature, °R	—	<806
Fuel Boost Pump Discharge Pressure, psia	47	74
Fuel Boost Pump Speed, rpm	30,600	24,681
Fuel Boost Pump Turbine Inlet Pressure, psia	—	4438
Fuel Boost Pump Turbine Inle Temperature, °R	—	<806
Oxidizer Pump Discharge Pressure, psia	2160	4852
Oxidizer Pump Speed, rpm	56,790	70,000
Oxidizer Pump Turbine Inlet Pressure, psia	1950	3297
Oxidizer Pump Turbine Inlet Temperature, °R	815	1860
Fuel Pump Discharge Pressure, psia	3390	4563
Fuel Pump Speed, rpm	115,000	95,000
Fuel Pump Turbine Inlet Pressure, psia	3440	3297
Fuel Pump Turbine Inlet Temperature, °R	884	1860
Preburner Combustion Pressure, psia	—	3307
Preburner Combustion Temperature, °R	—	186[illegible]

TABLE 6-5. COMPARISON OF ADVANCED-EXPANDER AND STAGED-COMBUSTION ENGINE DESIGN FEATURES

Component	*Advanced Expander*	*Staged Combustion*
Igniter	• Hydrogen-Cooled Torch Igniter • Two Spark Igniters and Exciters • No Igniter Propellant Valves	• Two Hydrogen-Cooled Torch Igniters (Preburner and Main Chamber) • Two Spark Igniters and Exciters for Preburner • Two Spark Igniters and Exciters for Main Chamber • Two Igniter Propellant Valves for Preburner • Two Igniter Propellant Valves for Main Chamber
Fuel Boost Pump	• Inducer Gear Driven by Fuel Pump Turbine • Two Hydrogen-Cooled Bearings • Sealed With Labyrinth Seals Upstream of Rear Bearing	• Inducer Driven by Separate 1-Stage Turbine • Honeycomb Tip Seal • Three Bearings
Oxidizer Boost Pump	• Inducer Gear Driven by Oxidizer Pump Turbine • Two Bearings - Front LO_2 Cooled and Rear LH_2 Cooled • Shaft Seal Package to Isolate GH_2 in Gear-Box from LO_2 in Pump	• Inducer Driven by Separate 1-Stage Turbine • Two Bearings • Three Shaft Seals to Separate Turbine GH_2 from LO_2 in Pump
Fuel Turbopump	• 2-Stage Centrifugal • Double-Acting Thrust Balance Piston • Back to Back Shrouded Impellers • 2-Stage Partial Admission Turbine Drive • Shrouded Blades With Blade Tip Labyrinth Seals • Two Bearings Hydrogen-Cooled	• 3-Stage Centrifugal • Balance Piston for Thrust Balance • External Seal Joints on Each Cross-Over for Access • 2-Stage Turbine Drive With Uncooled Blades • Shaft Seal Package Pressurized With LH to Isolate Pump from Turbine • Sheet Metal Liner in Inlet Manifold to Minimize Low Cycle Fatigue • Four Bearings

TABLE 6-5. COMPARISON OF ADVANCED-EXPANDER AND STAGED-COMBUSTION ENGINE DESIGN FEATURES (Continued)

Component	*Advanced Expander*	*Staged Combustion*
Oxidizer Turbopump	• 1-Stage Centrifugal With Shrouded Impeller • Single-Acting Thrust Piston • 1-Stage Full Admission Turbine Drive • Shrouded Turbine Blades • Shaft Seal Package to Isolate GH_2 in Gearbox from LO_2 in Pump • Turbine in Series With Fuel Pump Turbine • Two Bearings - Front LO_2 Cooled and Rear LH_2 Cooled • Speed Synchronized With Fuel Pump by Synchronizing Gear Between Two Pumps	• 1-Stage Centrifugal With Inducer • Balance Piston for Thrust Balance • 1-Stage Turbine Drive With Uncooled Blades • Three Shaft Seals With GHe Dam to Separate LO_2 from GH_2 • Sheet Metal Liner in Turbine Inlet Manifold to Minimize Low-Cycle Fatigue • Four Bearings - Front LO_2 Cooled and Rear LH_2 Cooled
Gearbox	• Common Gearbox for All Pumps • Two Power Drive Gear Trains • One Synchronizing Gear Train • GH_2-Cooled Gears With Dry Lubricant	• None
Preburner	• None	• Co-Axial Injector • GH_2 Cooled Liner
Combustion Chamber	• Milled Copper Channel Chamber - Area Ratio of 6:1 • 15-in. Chamber Length • Single Pass Parallel Flow Cooling	• Milled Copper Channel Chamber - Area Ratio of 14:1 • Single Pass Counter Flow Cooling
Regenerative Nozzle	• Tapered Tubes Furnace Brazed • 360 Tubes (180 Short and 180 Long) • Area Ratio of 208:1 • Two-Pass Cooling (Parallel, Then Counter Flow)	• Brazed Tubular Construction • 400 Tubes • Area Ratio of 175:1 • Two-Pass Cooling (Parallel, Then Counter Flow)
Extendible Nozzle	• Radiation Cooled - Carbon/Carbon Material • Area Ratio of 640:1 • No H_2 Cooling Required	• Dump Cooled With Cooling Flow from Fuel Pump Discharge • Area Ratio of 400:1 • Nozzles at End of Each Flow Tube on Nozzle • 1012 Round Cross-Section Tubes
Main Injector	• Co-Axial Elements • Regimesh Face With GH_2 Cooling • Tangential Slot Oxidizer Elements	• 108 Co-Axial Elements • Regimesh Face With GH_2 Cooling • Caps Partially Over Entrance of Oxidizer Elements Near Oxidizer Inlet to Given Even Flow Distribution Across Injector Face
GO_2 Heat Exchanger	• Milled Channel Plate Construction • Two Passes for LO_2; One Pass for GH_2 - Right Angle Flow • GO_2 Valve Controls Flow Through Hex • Provides GO_2 for T/P and THI	• Channel Wall Construction • Located Around Main Chamber Throat • Provides GO_2 for Igniters, T/P and THI
Regenerator	• Milled Channel Plate Construction • Single Pass for Both Flows - Right Angle Flow	• None

TABLE 6-5. COMPARISON OF ADVANCED-EXPANDER AND STAGED-COMBUSTION ENGINE DESIGN FEATURES (Continued)

Component	*Advanced Expander*	*Staged Combustion*
Extendible Nozzle Retraction Mechanism	• Uses Three Ballscrew Jackshafts • Two Drive Motors • Flexcable Between Jackshafts Maintains Synchronization • Cam Lock Secures Nozzle in Either Position • Electric Solenoid Used to Unlock Cam Lock	• Uses Three Rotating Screw Jacks • Driven by Single Power Source • Flexible Cable Used to Synchronize All Screw Jacks • Cam Lock Secures Nozzle in Position and Carries Load
Controls	• Four Solenoid Valves • Nine Propellant Valves (Two Make Up Main Fuel Control) • No Closed-Loop Controls • No Sensors Required for Control • No Modulating Control Valves • Electrical Power Required for Solenoids Only • Seven Propellant Valves Are Open/Closed Type and Two Schedule Areas • Operating Modes Are Selected by Actuating Various Combinations of Solenoids • Active Control Not Required Because All Pumps Are Geared Together	• 13 Propellant Valves • Uses Active Electronic Control System With Redundant Features • Electronic Actuators for Valves With Additional Pneumatic Actuators on Six Valves for Backup • Closed-Loop Control of Thrust and Mixture Ratio - Required to Maintain Safe Limits During Transients • Five Redline Parameters Monitored • Redundant Sensors Used for Major Control Parameters • All Valves Welded in Place to Minimize Leaks
Monitoring Instrumentation	• 13 Parameters Monitored	• 60 Parameters Monitored

TABLE 6-6. HAZARDOUS FAILURE CLASSIFICATIONS

Class 1:	Failure which is likely to cause death or complete system loss. Includes loss of all vehicle propellants or helium.
Class 2:	Failure which may cause injury or major system damage. Includes major damage to an engine.
Class 3:	Failure which may cause minor system damage resulting in a mission abort. Includes minor damage to an engine.

TABLE 6-7. FAILURE LIKELIHOOD CLASSIFICATIONS

Class 1:	High probability of occurrence at some time during life of engine.
Class 2:	Moderate probability of occurrence at some time during life of engine.
Class 3:	Low probability of occurrence at some time during life of engine.

TABLE 6-8. ADVANCED EXPANDER CYCLE FAILURE MODES

Item	*Function*	*Failure Types*	*Engine Condition At Time of Failure*	*Failure Effect On Engine*	*Failure Mode Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
Engine Gimbal	Provides a mount for the engine and permits gimbaling in a square vector pattern.	1. Structural failure	1. Steady-state	1. Thrust will not be supported; probable loss of propellant feed-system due to excessive vibration.	Yes	1	3
		2. Frozen pintle joint	2. Steady-state	2. Engine will not gimbal; thrust will not be affected.	No		
Ignition System	Furnishes a minimum of 20 sparks per second for engine ignition.	1. Fails to operate a. Arcing due to loss of pressure b. Shorting due to cracked insulation c. Circuit malfunction	1. Start	None. Dual igniters provide redundant system.	No		
Oxidizer Inlet Shutoff Valve (Normally Closed)	Seals off the oxidizer supply from the oxidizer pump; opens at initiation of the start signal and closes at shutdown.	1. Fails to open a. Jammed gear train, bearing or piston b. Ruptured bellows	1. THI	1. Engine will not operate due to loss of oxidizer supply; fuel will be lost overboard if fuel inlet shutoff valve is not reclosed.	No		
		2. Fails to remain open due to ruptured bellows	2. Acceleration and steady-state	2 Engine will shut down due to loss of oxidizer supply. Fuel will be lost overboard until fuel inlet shutoff valve is closed.	No		
		3. Fails to close a. Inability to remove helium signal b. Jammed gear train, bearing or piston	3. Shutdown	3. Engine will be shut down by the main fuel shutoff valve; vehicle prevalve will prevent any significant propellant loss.	No		
		4. Internal leakage	4. Coast	4. Vehicle prevalve will prevent any significant propellant loss.	No		

TABLE 6-8. ADVANCED EXPANDER CYCLE FAILURE MODES (Continued)

Item	*Function*	*Failure Types*	*Engine Condition At Time of Failure*	*Failure Effect On Engine*	*Failure Mode Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
Fuel Inlet Shutoff Valve (Normally Closed)	Seals off the fuel supply from the fuel pump; opens at initiation of the start signal and closes at shutdown.	1. Fails to open a. Jammed gear train, bearing or piston b. Ruptured bellows	1. THI	1. Engine will not operate due to loss of fuel supply; oxidizer will be lost overboard if oxidizer inlet shutoff valve is not reclosed.	No		
		2. Fails to remain open due to ruptured bellows	2. Acceleration and steady-state	2. Engine will be shut down due to loss of fuel supply; oxidizer will be lost overboard until oxidizer inlet shutoff valve is closed.	No		
		3. Fails to close a. Inability to remove helium signal b. Jammed gear train, bearing or piston	3. Shutdown	3. Engine will be shut down by the fuel shutoff valve, vehicle prevalve will prevent any significant propellant loss.	No		
		4. Internal leakage	4. Coast	4. Vehicle prevalve will prevent any significant propellant loss.	No		
Start Solenoid Valve (Normally Closed)	Controls the THI start cycle by porting helium supply pressure to the fuel and oxidizer inlet shutoff valves and fuel vent valves; also supplies helium to the other solenoid valves.	1. Fails to open a. Burned out solenoid coil b. Shaft binding on guides	1. THI	1. Inlet valves will not open; engine will not operate	No		
		2. Fails to remain open due to burned out solenoid coil	2. Acceleration and steady-state	2. Inlet valves will close; engine will shut down.	No		
		3. Fails to close due to shaft binding on guides	3. Shutdown	3. Inlet valves will remain open; engine will be shut down by fuel shutoff valve. Vehicle prevalves will prevent any significant propellant loss	No		

TABLE 6-8. ADVANCED EXPANDER CYCLE FAILURE MODES (Continued)

Item	*Function*	*Failure Types*	*Engine Condition At Time of Failure*	*Failure Effect On Engine*	*Failure Mode Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
		4. Internal Leakage	4. Coast, all engine operating modes	4. Engine will shut down	No		
Fuel SOV Solenoid (Normally Closed)	Controls engine acceleration by porting helium supply pressure to the fuel shutoff valve	1. Fails to open a. Burned out solenoid coil b. Shaft binding on guides	1. Acceleration to pumped idle	1. Main fuel shutoff valve will not open; engine will not operate except in THI.	No		
		2. Fails to remain open due to burned out solenoid coil	2. Acceleration and steady-state	2. Engine will shut down to THI since main fuel shutoff valve will close.	No		
		3. Fails to close due to shaft binding on guides	3. Shutdown	3. Engine will shut down since helium pressure will be vented through the start solenoid valve.	No		
Bypass Solenoid Valve No. 1	Routes helium pressure to turbine bypass valve to open bypass for THI Removal of helium pressure at pumped idle permits turbine bypass valve to close to full thrust position.	1. Fails to open a. Burned out solenoid coil b. Shaft binding on guides	1. Start	1. Engine will not operate due to fuel not flowing through bypass valve for THI Oxygen will be lost through injector.	No		
		2. Fails to remain open due to burned out solenoid coil.	2. Acceleration and steady-state (pumped idle)	2. Turbine bypass valve will close to full thrust position. Engine will accelerate to full thrust.			
		3. Fails to close due to shaft binding on guides	3. a. Acceleration and steady-state (full thurst)	3.a. Engine will decel to pumped idle	No		

TABLE 6-8. ADVANCED EXPANDER CYCLE FAILURE MODES (Continued)

Item	*Function*	*Failure Types*	*Engine Condition At Time of Failure*	*Failure Effect On Engine*	*Failure Mode Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
			b. Shutdown	b. Engine will shut down normally as helium will be vented through start solenoid.	No		
Bypass Solenoid Valve No. 2	Routes turbine inlet pressure to turbine bypass valve to close bypass during acceleration to pumped idle	1. Fails to open a. Burned out solenoid coil b. Shaft binding on guides	1. Acceleration to pumped inlet	1. Engine will remain in THI.	No		
		2. Fails to remain open due to burned out solenoid coil	2. Acceleration and steady-state (pumped idle)	2. Engine will decel to THI.	No		
		3. Fails to close due to shaft binding on guides.	3. Shutdown	3. Engine will shut down normally. Turbine inlet pressure will decay after fuel SOV and inlet valve close.	No		
		4. Internal leakage	4. Acceleration and steady-state	4. Helium will be lost overboard while start solenoid is energized.	No		
Fuel Shutoff Valve (Normally Closed)	Restricts turbine flow during THI and shuts off fuel through injector at shutdown	1. Fails to open a. Binding piston or actuating rod b. Loss of helium signal c. Ruptured bellows	1. Acceleration and steady-state (pumped idle)	1. Engine will not accel from THI since fuel flow through turbine is prevented	No		
		2. Fails to remain open a. Loss of helium signal b. Ruptured bellows c. Excessive helium leakage	2. Steady-state (full thrust)	2. Engine will shut down due to interruption of fuel flow to the injector; fuel pump and inlet line may experience a high pressure surge; oxidizer will be lost	Possibly	2	3

TABLE 6-8. ADVANCED EXPANDER CYCLE FAILURE MODES (Continued)

Item	*Function*	*Failure Types*	*Engine Condition At Time of Failure*	*Failure Effect On Engine*	*Failure Mode Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
		3. Fails to close a. Jammed piston or actuation rod b. Unable to remove helium signal	3. Shutdown	3. Engine will be shut down by the inlet shutoff valves; shutdown impulse will be abnormally high.	No		
		4. Minor leakage past valve	4. THI	4. Some cooling of turbine	No		
Oxidizer Flow Control	Controls oxidizer flow during pumped idle and full thrust engine operation for proper mixture ratio	1. Fails to open due to binding oxidizer flow control inlet valve	1. Acceleration to pumped inlet	1. Engine will not accelerate	No		
		2. Fails to close due to binding oxidizer flow control inlet valve	2. Shutdown	2. Engine shutdown will not be adversely affected.	No		
		3. Leakage High-pressure leak past piston to oxidizer inlet pressure	3. Steady-state	3. No effect on engine operation	No		

TABLE 6-8. ADVANCED EXPANDER CYCLE FAILURE MODES (Continued)

Item	*Function*	*Failure Types*	*Engine Condition At Time of Failure*	*Failure Effect On Engine*	*Failure Mode Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
Main Fuel Control (Turbine Bypass Valve)	Schedule turbine bypass area to bypass all flow during THI and set required bypass for pumped idle and full thrust.	1. Fails to move from full closed position	1. Start to THI	1. Engine will not start	No		
		2. Fails to open fully to THI position (full open)	2. Start to THI	2. Engine may not start if sufficient fuel can not pass through valve. Fuel will be lost overboard.	No		
		3. Fails to close to pumped idle position	3. Acceleration to pumped idle	3. Engine will accelerate to some lower thrust level	No		
		4. Fails to close to full thrust position	4. Acceleration to full thrust	4. Engine will accelerate to some lower thrust level	No		
		5. Fails to go fully closed	5. Shutdown	5. Fuel will be shutoff with fuel inlet valve.	No		
Main Fuel Control (Vent Valve)	Vents trapped fuel overboard at shutdown	1. Fails to close fully	1. Start to THI	1. Engine may not start if too much fuel is being vented overboard. Fuel will be lost overboard.	No		
		2. Fails to open	2. Shutdown	2. Improper shutdown; fuel system pressure will not be relieved. Fuel pumps will experience a high pressure surge.	Possibly	2	2
GO2 Valve	Meters gaseous oxygen flow to injector during THI and acceleration to pumped idle	1. Fails to close properly	1. Acceleration to pumped idle; steady-state	1. Mixture ratio will be too high; some thrust chamber and injector overtemperature may occur.	Possibly	3	2

TABLE 6-8. ADVANCED EXPANDER CYCLE FAILURE MODES (Continued)

Item	*Function*	*Failure Types*	*Engine Condition At Time of Failure*	*Failure Effect On Engine*	*Failure Mode Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
		2. Fails to open	2. Shutdown	2. No effect on shutdown. Engine will not start during next THI.	No		
Tank Pressurization Valves (Oxidizer and Fuel)	Permits tank pressurization flow to be extracted from engine at pumped idle and above. Prohibits propellant backflow from tanks during THI	1. Fails to open	1. Acceleration to pumped idle	1. Propellant tank will not be pressurized by engine	No		
		2. Fails to close	2. Shutdown	2. The vehicle system valve would prevent significant propellant loss.	No		
Propellant Piping	Transports propellants - rigid tubes greater than 1.000 outer diameter with flanges of the conical crushed seal type.	1. Tube rupture	1. Start and steady-state	1. Engine will shut down.	Yes	2	3
		2. Leakage	2. THI, acceleration, steady-state	2. Propellants will be lost overboard; excessive leakage may cause engine shutdown.	Possibly, if excessive	3	1
Engine Plumbing	Transport helium to controls and valves, transports propellants and chamber pressure, 3/16 in. and 1/4 in. lines, metal-to-metal conical seal connections	1. Tube rupture - helium	1. THI, acceleration and steady-state	1. Engine will shut down	Yes	3	3
		2. Tube rupture - propellant	2. THI, acceleration and steady-state	2. Propellants will be lost overboard, engine may shut down	Yes	3	3
		3. Leakage	3. THI, acceleration and steady-state	3. Helium or propellants will be lost overboard	Possibly, if excessive	3	1

TABLE 6-8. ADVANCED EXPANDER CYCLE FAILURE MODES (Continued)

Item	*Function*	*Failure Types*	*Engine Condition At Time of Failure*	*Failure Effect On Engine*	*Failure Mode Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
Propellant Injector	Atomizes and mixes propellants for efficient combustion	1. Structural failure a. Fuel leakage to outside b. Oxidizer leakage to outside	THI, acceleration and steady-state	1. Propellants will be lost overboard	Yes	3	3
		2. Plugged propellant passages a. Oxidizer nozzles b. Fuel transpiration surface	2. THI, acceleration and steady-state	2. Engine will lose thrust if a large number of oxidizer nozzles are plugged. Local overheating will occur if an excessive amount of the transpiration surface is plugged.	No		
Thrust Chamber	Provides a chamber where propellants react, gaseous products are expanded through a nozzle to provide thust; also provides a heat exchanger for operation of the turbine.	1. Failed component attachment bracket	1. THI, acceleration and steady-state	1. Component not supported; probable propellant piping failure	Possibly	3	3
		2. Failed actuator mounts	2. THI, acceleration and steady-state	2. Engine will not gimbal in the correct pattern.	No		
		3. Burst tubes	3. THI, acceleration and steady state	3. Turbine will lose power; engine may shut down	Yes	3	2
		4. Chamber distortion	4. THI, acceleration and steady-state	4. Thrust may be slightly reduced.	No		
		5. Fuel passage leak	5. THI, acceleration and steady-state	5. Fuel will be lost overboard.	No		
		6. Leak through braze joint	6. THI, acceleration and	6. Engine will lose thrust if leak is excessive.	No		

TABLE 6-8. ADVANCED EXPANDER CYCLE FAILURE MODES (Continued)

Item	*Function*	*Failure Types*	*Engine Condition At Time of Failure*	*Failure Effect On Engine*	*Failure Mode Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
Turbopump Assembly	Pressurizes propellants by means of turbine driven pumps						
Fuel Low Speed Inducer		1. Failed gears — low speed induced gears, oxidizer pump gear, idler gears, fuel pump gear	1. THI, acceleration and steady state	1. Engine will shut down; propellants will be lost overboard	Yes	3	2
Oxidizer Low Speed Inducer		2. Ruptured housing	2. THI, acceleration and steady-state	2. Engine will shut down; propellants will be lost overboard	Yes	3	
Fuel Pump		3. Failed bearing	3. THI, acceleration and steady-state	3. Engine will shut down; propellants will be lost overboard	Yes	3	2
Oxidizer Pump Gearbox Turbines		4. Pump impeller tip failure	4. THI, acceleration and steady-state	4. Engine thrust reduced due to reduced pump performance; possible thrust chamber damage from blocked tubes	No		
		5. Turbine failure	5. THI, acceleration and steady state	5. Engine will shut down; propellants will be lost overboard	Yes	3	3
		6. H_2 seals, excessive leakage	6. THI, acceleration and steady state	6. Slight loss of pump performance, possible loss of propellants overboard	No		
		7. O_2 seals, excessive leakage	7. THI, acceleration and steady state	7. Slight loss of pump performance; possible loss of propellants overboard; evacuated cavities prevent mixing of propellants	No		
		8. Housing flange leakage	8. THI, acceleration and steady state	8. Propellants will be lost overboard	No		

TABLE 6-8. ADVANCED EXPANDER CYCLE FAILURE MODES (Continued)

Item	*Function*	*Failure Types*	*Engine Condition At Time of Failure*	*Failure Effect On Engine*	*Failure Mode Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
Extendible Nozzle	Provides additional area ratio for increased performance	1. Nozzle will not translate due to jammed translation system	1.a. Before initial start	1.a. None. Mission will be scrubbed.	No		
			1.b. After final burn	1.b. None. OTV cannot be placed in shuttle payload bay	No		
		2. Nozzle distortion	2. THI, acceleration and Steady-State	2. Slight loss in thrust and Isp.	No		
		3. Hole in nozzle due to burning	3. THI, acceleration and Steady-State	3. Some loss in performance	No		

*Notes:

1. This analysis is based on a single component failure and considers no secondary failure effects.
2. All component failures are considered to be complete. Degrees of failure are not considered.

Design features and operating characteristics for the staged combustion engine were then compared to those of the Advanced Expander engine to identify relative failure modes. The relative comparison was made because sufficient detailed information was not available for the staged combustion engine to accomplish a complete FMEA. Each component was compared for both engines to make the failure mode comparison. The relative comparison of failure modes is shown in Tables 6-9 and 6-10. The staged combustion engine has at least 49 more failure modes that are unique to it while the Advanced Expander has 6 that are unique. This results in the staged combustion engine having a total of at least 43 more failure modes than the Advanced Expander engine.

TABLE 6-9. COMPARISON OF POTENTIAL FAILURE MODES FOR STAGED-COMBUSTION ENGINE RELATIVE TO ADVANCED-EXPANDER ENGINE

Component	*Failure Mode Comparison*	*Failure Modes Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
Igniter System	Same failure modes as Advanced Expander Engine except:			
	1. More potential failures - Two complete systems used instead of one.	Some can be	3	3 (many systems have redundancy)
	2. Igniter propellant valves may fail to operate.			
	a. Fail to open (no ignition)	No		
	b. Fail to close (may burn igniter)	Yes	3	3
Fuel Boost Pump	Same failure modes as Advanced Expander Engine except:			
	1. Potential for one additional bearing to fail	Probably not		
Oxidizer Boost Pump	Same failure modes as Advanced Expander Engine	—		
Fuel Turbopump	Same failure modes as Advanced Expander Engine except:			
	1. Potential for two additional bearings to fail.	Yes	2	2
	2. Potential for turbine failure due to overtemperature	Yes	1	1
	3. More potential for failures associated with impeller rub - Uses three stages instead of two.	Yes	3	3
	4. More potential for failures related to seals. Much larger temperature difference between pump and turbine fluids. LH_2 required in seal package.	Yes	2	2
	5. Some potential for sheetmetal liner in turbine inlet manifold and turbine disk cover to fail. Not needed on lower temperature expander engine.	No		

TABLE 6-9. COMPARISON OF POTENTIAL FAILURE MODES FOR STAGED-COMBUSTION ENGINE RELATIVE TO ADVANCED-EXPANDER ENGINE (Continued)

Component	*Failure Mode Comparison*	*Failure Modes Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
Oxidizer Turbopump	Same failure modes as Advanced Expander Engine except:			
	1. Potential for two additional bearing failures.	Yes	2	2
	2. Potential for turbine failure due to overtemperature	Yes	1	1
	3. More potential for failures related to seals.			
	a. Much larger temperature difference between pump and turbine fluids.	Yes	1	2
	b. Potential for failure of GHe supply to seal package.	Yes	1	3
	4. Some potential for sheetmetal liner in turbine inlet manifold to fail. Not needed on lower temperature expander engine.	No		
Preburner	All failure modes are unique to Staged Combustion Engine since preburner not required for expander engine. Some preburner failure modes which could occur are:			
	1. Non-uniform temperature profile could cause turbine distress and failure.	Yes	1	2
	2. Non-uniform liner cooling could cause burnout of preburner combustor.	No		
Main Combustion Chamber	Same failure modes as Advanced Expander Engine except:			
	1. Heat flux higher more potential for chamber burnout.	Probably Not		
Regenerative Nozzle	Same failure modes as Advanced Expander Engine except:			
	1. Heat flux higher more potential for nozzle tube burnout	Probably Not		
Extendible Nozzle	More potential failure modes with dump cooled nozzle.			
	1. Potential for tube burnout due to blockage or poor distribution of cooling flow	Probably Not		

TABLE 6-9. COMPARISON OF POTENTIAL FAILURE MODES FOR STAGED-COMBUSTION ENGINE RELATIVE TO ADVANCED-EXPANDER ENGINE (Continued)

Component	*Failure Mode Comparison*	*Failure Modes Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
Main Injector	Same failure modes as Advanced Expander Engine.			
GO_2 Heat Exchanger	Same failure modes as Advanced Expander Engine except:			
	1. More potential for failures because two valves (fuel shunt and GO_2 flow control) are used to control heat exchanger flows.	No		
Extendible Nozzle Retraction Mechanism	Same failure modes as Advanced- Expander Engine.			
Controls	Many more potential failure modes in control system:			
	1. Potential for four more propellant valve failures (13 vs 9 valves used)	Yes	1	2
	2. Potential for failure of actuators on all valves.	Yes	1	3 (redundancy on major valves)
	3. Potential for misscheduling of three modulating valves. Critical during transients.	Yes	1	1
	4. Potential for many failures in electronic control system and associated electrical equipment.	Yes	1	2
	5. Potential for failure of closed-loop control sensors.	Yes	1	3
	6. Potential for failure of five redline parameter sensors.	Yes	1	3
Monitoring Instrumentation	Same failure modes as Advanced Expander Engine except potential for many more (60 vs 13 parameters monitored).	No		

These relative failure modes were evaluated to ascertain if they would be hazardous, and classified for severity and likelihood using the same criteria as used earlier for the Advanced Expander engine. This information is included in Tables 6-9 and 6-10 for each failure mode. A total of [illegible] of the failure modes unique to the staged combustion engine were found to have No. 1 or No. 2 hazardous classifications. One was found in the 6 failure modes unique to the expander cycle engine. Based on this information it was estimated that 30% of the failures for a staged combustion engine would result in damage to an adjacent engine. It was also estimated that at least 15% would result in damage to the vehicle. This compares to 6 and 3%, respectively estimated for the Advanced Expander engine. The number of failure modes estimated for the staged combustion engine is probably low since sufficient information was not available to make a detailed evaluation of the control system. However, many more failure modes than indicated probably exist in that area.

TABLE 6-10. EVALUATION OF POTENTIAL FAILURE MODES UNIQUE TO ADVANCED-EXPANDER ENGINE

Component	*Failure Mode*	*Failure Modes Hazardous*	*Hazardous Failure Classification*	*Failure Likelihood Classification*
Gearbox	Failed gear	Can be	3	3
Controls	Failed solenoid valve - Four used on Advanced Expander Cycle Engine.			
	1. Start solenoid fails open at shutdown. Inlet valves will remain open; engine will be shut down by fuel shutoff valve. Vehicle prevalves will prevent any significant propellant loss.	No		
	2. Start solenoid has internal leakage during all operating modes and coast producing unscheduled valve actuation.	Possibly, if leakage high	1	3
	3. Failure of other solenoid valves	No		

6.5 MISSION AND CREW SAFETY RELIABILITY COMPARISON

To compare mission and crew safety reliabilities for the staged-combustion and advanced-expander cycle engines, it was necessary that estimates of both engine reliability and the percentage of failures that could damage the vehicle or an adjacent engine be made for both engine configurations. The damage to adjacent engines was needed to determine crew safety reliability on multiple engine vehicles where the crew can return safely using a backup propulsion system or with one of two engines out.

Although the Advanced-Expander engine has more valves than the current RL10, elimination of the closed-loop thrust control and propellant utilization mixture ratio control makes the total number of failure modes nearly the same as the RL10A-3-3. The current demonstrated 90% confidence reliability for the RL10A-3-3 engine is 0.9982 (based on 1431 accountable firings). Since an engine's reliability is related to the number of potential failure modes and engine configuration (e.g., control system, cycle, etc.) it was assumed that the advanced-expander cycle would have essentially the same percentage of accountable firings at FFC as did the RL10A-3-3. Based on a total of 1250 firings for the advanced-expander cycle at FFC, it is estimated that 700 firings would be accountable for reliability purposes.

The staged-combustion engine configuration (from Contract NAS8-32996) is similar to the SSME (e.g. control system, cycle, etc.) and is expected to have a similar number of failure modes. The projected percentage of accountable to total firings at FPLC for the SSME (22.5%) was applied to the estimated total 1000 engine tests at FFC were estimated under Contract NAS8-32996) to be accomplished on a staged combustion OTV engine. This results in an estimated 225 accountable firings at FFC.

Using these assumptions, demonstrated engine single-burn reliability at FFC for the advanced-expander cycle was estimated to be 0.9967 and for the staged-combustion engine was estimated to be 0.9898.

These values are assumed reasonable reliability estimates for the two engine configurations, based on the information available.

During the failure mode comparison, estimates were made of the percentage of engine failures that would disable the spacecraft or adjacent engines for both the staged-combustion and advanced-expander engines. New parametric curves of crew safety reliability as a function of engine burn reliability were generated using these values. New curves were generated for the two engines with one engine out, and one engine with APS backup OTV vehicles only, since the previous parametric curves had shown that more than one engine is needed to provide a reasonable crew safety reliability level. These curves are shown in Figure 6-8. These engine reliability values are estimated on the curves. For reference, mission reliability curves are also included with the engine reliability values shown on them.

The following crew safety and mission reliability values result for the advanced-expander and staged-combustion engine configurations:

	Advanced Expander (0.9967 Engine Reliability)		*Staged Combustion (0.9898 Engine Reliability)*	
Configuration	*Mission Reliability*	*Crew Safety Reliability*	*Mission Reliability*	*Crew Safety Reliability*
1 Main Engine	0.9804	0.9901	0.9403	0.9697
With APS Backup for Crew Safety	0.9804	0.9996	0.9403	0.9945
2 Main Engines With Engine-Out Capability for Crew Safety	0.9611	0.9987	0.8842	0.9809

These data indicate that both crew safety and mission reliability will be significantly higher with the advanced-expander engine than with the staged-combustion engine. While these absolute levels may not be exact, the relative levels and trends should be indicative of the differences that exist.

Configuration	Mission	Safety	Mission Reliability	Safety Reliability
1 Main Engine, 1 APS	1 ME	1 ME or APS	M2	S2
2 Main Engines	2 ME	1 ME	M3	S3

Assumptions:

- 30% of main engine failures for staged combustion cycle destroy adjacent engine but not APS
- 6% of main engine failures for advanced expander cycle destroy adjacent engine but not APS
- 15% of main engine failures for staged combustion cycle disables spacecraft (APS)
- 3% of main engine failures for advanced expander cycle disables spacecraft (APS)
- APS is equally reliable as main engine
- No rescue facility
- 6 burns/mission - 3 burns/safety mission

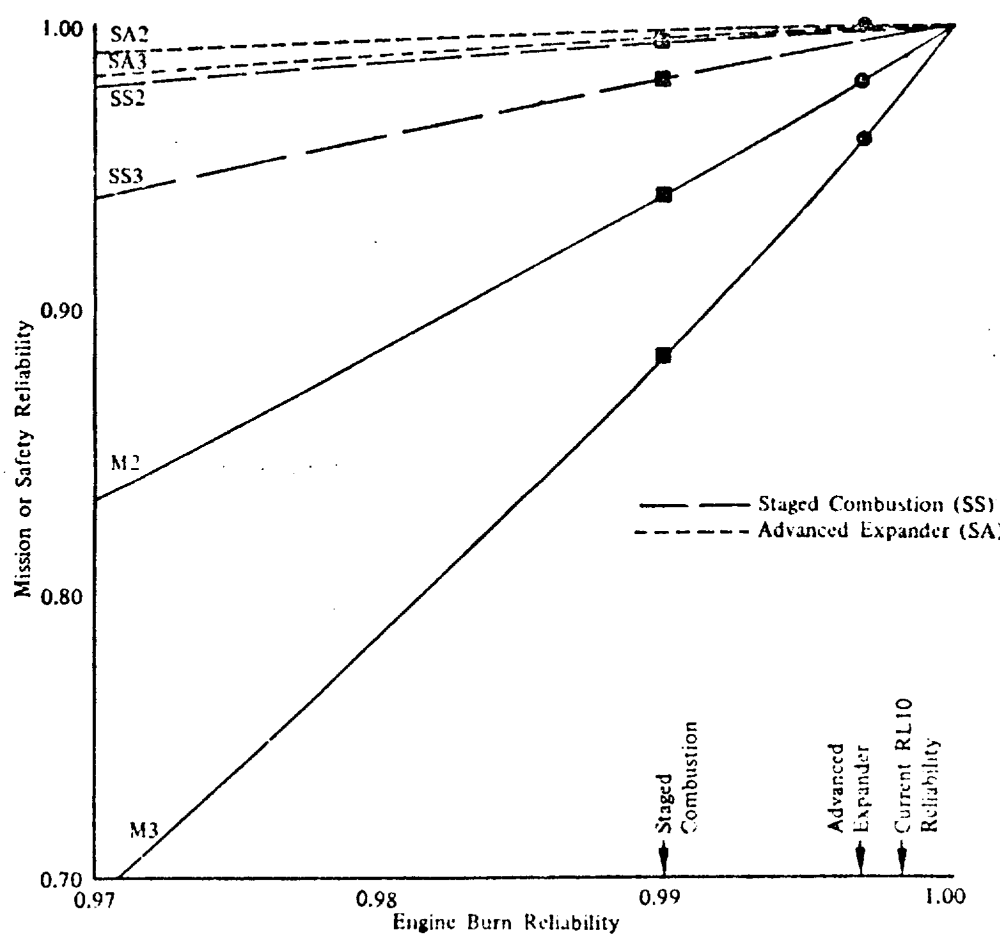

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Figure 6-8. OTV Reliability With Advanced Engines

SECTION 7

PROGRAM PLANS

7.0 SUMMARY

Development plans established for modified 15,000-lb thrust RL10A-3-3 engines (Derivatives IIA and IIB) and optimized expander cycle RL10 engine (Category IV) in the 1973 Contract NAS8-28989 "Design Study of RL10 Derivatives" were used as the basis for the plans presented here. Plans were adjusted to reflect the procurement lead times currently being experienced, and any new information available. A program plan for the new 15K-lb thrust Advanced Expander Cycle Engine was generated during this study.

The engine development program approach used in the program planning for the Contract NAS8-28989 study was based on design verification specifications (DVS) which specify the design requirements and method of verifying these requirements for the baseline RL10 Derivative engines (Derivatives IIA and IIB). DVS's were not generated for the Category IV engine, but an estimate of the verification program for this engine was made. The Advanced Expander engine was estimated in a similar manner. The DVS's establish a minimum development program because the assumption is made that the development program is "success oriented," and only one design, build, test cycle through engine Final Flight Certification is required. Knowing that previous RL10 and other rocket engine (e.g., F-1 and J-2) development programs have not been accomplished in a single cycle, a redesign and reverification effort has been considered in the total engine development program plans. The total development program effort planned for each engine design was based on data and experience from previous RL10 engine programs. The redesign and reverification effort was determined by estimating the DVS requirements and deducting these from the total engine development program requirements.

Preliminary program plans were developed for each baseline engine design configuration for the total development through Final Flight Certification (FFC). Program planning was based on DVS's formulated for those RL10 engine components not already qualified, i.e., components that are not of the same configuration as those used in the operational RL10A-3-3 engine that is currently used in the Centaur launch vehicle. As stated above, a redesign and reverification effort was included to achieve a realistic total engine development program. The major milestones and key decision points, as well as other significant activities of these programs, were derived and the durations established for the specified tasks. The number of hardware components and engines required in equivalent engine sets, and the number of engine tests were specified for both the DVS program requirements and the total development program requirements.

Test facilities required for engine development and Ground Support Equipment development were identified. Other end items, including packaging, preservation, handling and mockup activities were also specified.

Budgetary and planning cost estimates for each baseline RL10 engine Category (Derivative IIA, IIB, IIC, Category IV and Advanced Expander) are presented in Volume III of this Report. These cost estimates were determined for the development engine program, the first production unit, and the Operational and Flight Support programs.

7.1 ENGINE DEVELOPMENT PROGRAM APPROACH

7.1.1 General

The engine development programs for the RL10 Derivative (Derivatives IIA, IIB, II Category IV, and Advanced Expander Cycle engines considered in this and earlier study a preliminary in nature and encompass development through flight certification, achievement engine readiness for production, and field operations starting from current operation RL10A-3-3 engine technology levels. The programs are directed toward minimizing the r (through use of previous RL10 engine development program data and experience) involved any major development effort by verifying that the engine designs meet the limits of t requirements at the lowest hardware level, and identifying marginal conditions by conducti selective overstress tests at detail, subcomponent, component, subsystem and engine levels

In formulating the programs, emphasis was placed on making a well integrated effort verify the new design requirements, operational modes, and ground support equipment of RL10-derived engines. An essential part of this development effort was the early verification potential failure modes and preparation of detailed contingency plans to ensure early probl solutions and thus, minimize impact on the program. Changes in the baseline design conc were made only for the resolution of problems and for verifying that the new design features r the specified requirements at the lowest practical hardware assembly level as early in program as possible. Marginal improvements were excluded.

High hardware and facility costs are risked in overstress testing because of possi catastrophic failures that may be induced, however, selective overstress testing can significar reduce engine development cost and time by accelerating the learning rate. Subcompon overstress testing (material specimen, rotating part spin, housing pressure, etc.) is u extensively by P&WA to find and correct problems at the lowest possible hardware level, there avoiding risks of damage to more expensive hardware. In addition to these subcomponent te selective component, subsystem, and engine overstress tests should be integrated into the ove engine development programs as part of the design verification requirement. These overstress t requirements are specified in engine and component DVS's.

Satisfactory verification that all requirements have been met is established by a for flight certification demonstration conducted under the cognizance of the procuring agen Certification of the service life capability of the design is accomplished based on the cumulat life history of all the development hardware, and a final life demonstration conducted usi simulated typical mission duty cycle.

Verification is based on (1) demonstration that the components and engines can perfo within the margins intended in the design when the engine is operating over its range of ra conditions, and (2) demonstration that the engine has a level of operational maturity, free failures to the degree necessary for safety when installed in the OTV stage. When these go are achieved the engine is ready for flight, where service life can be determined by continu use with frequent inspections until a cause for removal from the vehicle is found. As th causes are corrected, service life will grow, the rate of growth depending mostly on the rate accumulating flight experience, until the target of 190 firings and 5 hr of Derivative IIA a IIB engines, and 300 firings and 10 hr for Category IV and Advanced Expander engines achieved. With further experience, these targets should be passed in a continuing growth service life Time Between Refurbishment (TBR).

Implementing this approach to flight certification of long life reusable hardware results in the establishment of four goals:

1. All Design Verification Specifications (DVS's) must have been satisfactorily completed to the applicable certification level (either PFC or FFC).

2. A level of maturity must be demonstrated in the engine development program that provides safety for the intended period of operation between inspections.

3. Completion of required design reviews and configuration inspections.

4. Demonstration of engine service and subsequent teardown and hardware inspections (PFC and FFC).

7.1.2 Design Verification and Design Verification Control

For the engine development programs considered in this study, the key to the process is the DVS. These specifications identify the design requirements to be verified and the method of verification of these requirements for each component not previously qualified for flight operation.

The DVS forms the foundation for the Program Development Plans. During the engine development, failure modes are identified and eliminated at the lowest practical level. Detail parts, subassemblies, and minor components are subjected to overstress testing and tested to failure early in the program. Initial component test emphasis is to verify components for engine testing and to verify performance parameters at the component level. Component test emphasis then shifts as quickly as practical to tests designed to identify failure modes. The engine tests are planned so that all DVS test objectives are considered on each test. Using this information as a base, an engineering estimate of the total number of tests and the required supporting hardware was made for DVS testing. The estimates reflect the degree of difficulty and the amount of directly related experience applied to that design. Section 7.3 covers this subject and shows the results obtained for the OTV engine development through FFC.

7.1.3 Verification Method

The DVS verification method identified in the DVS's for each design requirement specifies verification accomplishment at five levels. These are: analysis, hardware inspection, laboratory or bench tests, subsystems hot-fire tests, and engine system hot-fire tests. The DVS's further specify all tests at all levels that are estimated to be required for verification of the design requirements. The testing levels referred to are detail part, subcomponent, component, subsystem, and engine.

7.2 ENGINE DEVELOPMENT PROGRAMS

7.2.1 Summary

Preliminary program plans had been formulated for the complete engine development program through final flight certification (FFC) for FL10 engines Derivative IIA, Derivative IIB and Category IV in the earlier study. A plan based upon these was developed for the Derivative IIC and Advanced Expander Cycle Engines. Program planning was based on previous RL10 engine development program data and experience.

The unique feature of the Derivative IIA and IIB engine program plans is the Design Verification Specification (DVS) approach to component and engine development. Ten preliminary DVS's, which define the design requirements and method of verification of these requirements, were prepared for the De ivative IIA engine development programs. Nine of the DVS's apply to the Derivative IIB engine development program because the Derivative IIB engine design is like the Derivative IIA engine design, with the exception that the IIB incorporates the RL10A-3-3 engine parts list turbomachinery and propellant inlet shutoff valves. These preliminary DVS's form the foundation of the engine development programs. Each DVS defines the detailed hardware and test requirements necessary to verify a single design. There are some "to-be-determined" (TBD) requirements in the DVS's which require information from CEI Specifications.

A limited distribution of these Design Verification Specifications (generated under Contract NAS8-28989) was made including copies to:

Defense D cumentation Center Headquarters
TISIA
Cameron Station, Building 5,
5010 Duke Street,
Alexandria, Virginia 22314

and

NASA Scientific and Technical Information Facility,
P. O. Box 33
College Park, Maryland 20740

The program planning effort for the Derivative IIC, Category IV and Advanced Expander Cycle engines did not include formulation of DVS's. However, definitions of the anticipated DVS test requirements to accomplish these engines' development program objectives are provided.

7.2.2 Derivative IIA Engine

The baseline Derivative IIA engine design is a basic RL10A-3-3 modified to best satisfy the OTV operating requirements by incorporating concepts that have already been demonstrated. The Derivative IIA engine consists of an RL10A-3-3 engine with:

1. Recontoured, high expansion ratio, two position nozzle

2. Reoptimized injector

3. Tank head idle mode capability

4. "Zero" NPSH operation capability

5. Low thrust capability (defined as the lowest powered stable operation level achievable without significant design impacts) and herein designated maneuvering thrust

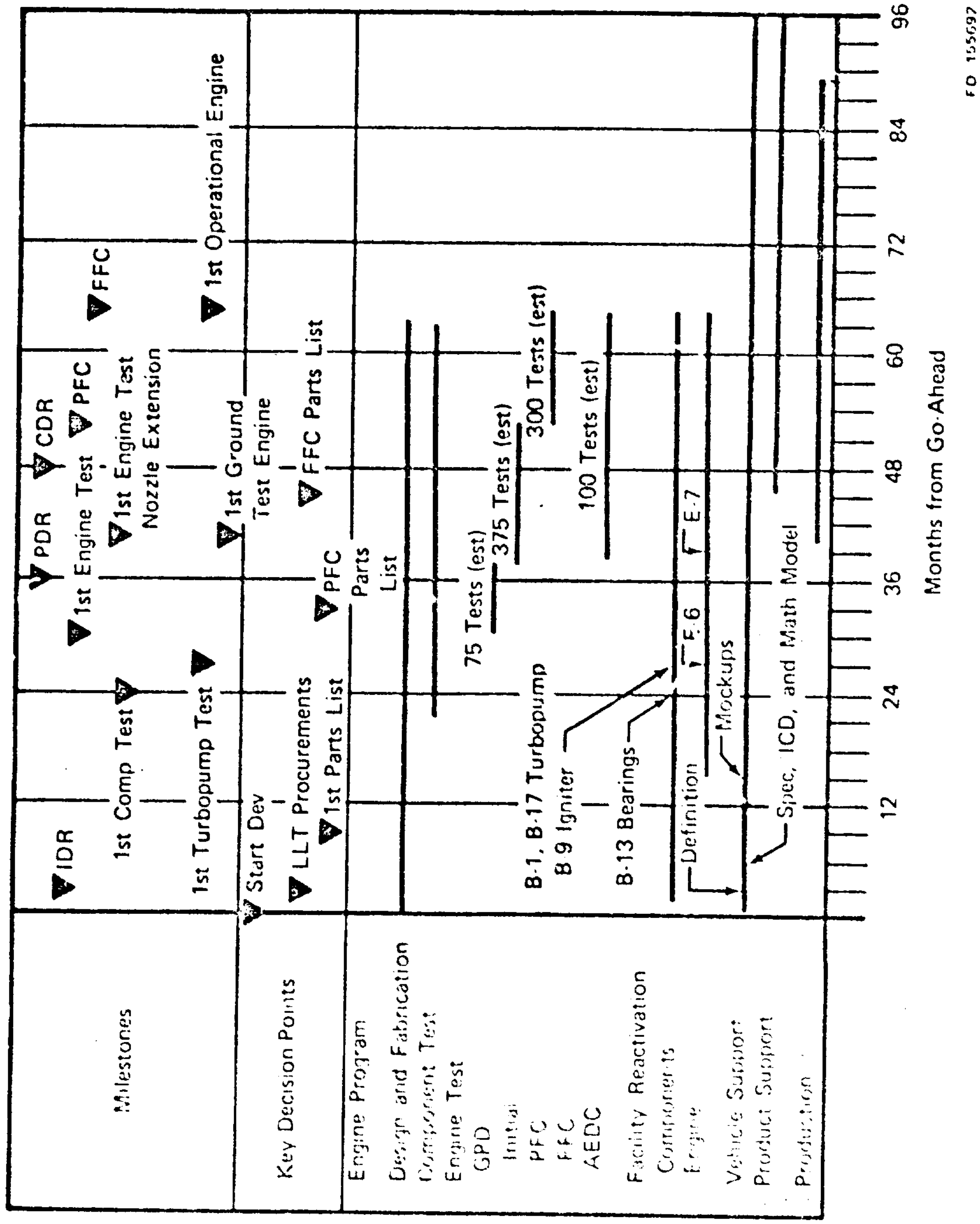
6. Autogenous pressurization capability (fuel and oxidizer).

The total engine development program for the baseline Derivative IIA engine will require about 64 months of design, fabrication, and test effort. This effort will encompass three design, build, test cycles to FFC (Initial, PFC and FFC configurations). Figure 7-1 depicts the development schedule, presenting the major program milestones and key decision points as well as the total engine development program. The design and fabrication schedules for this program are shown in Figure 7-2, and the total program test plan is shown in Figure 7-3.

The design and fabrication schedules are planned for early release of long-lead-time material procurement as well as sequential releases of drawings to support the fabrication process and permit meeting the early hardware delivery schedules. The major design effort will be placed on the design of the "all new" components for the Derivative IIA engine which include a recontoured thrust chamber, extendible nozzle, extendible nozzle translating mechanism and coolant feed systems, oxidizer low speed inducer and drive system, GOX heat exchanger, turbine bypass valves, tank pressurizing valves, and gaseous oxidizer valve. The remaining design effort is placed on modification of existing RL10A-3-3 components to incorporate Derivative IIA engine peculiar operating features. These components include fuel pump and turbine, oxidizer pump, propellant inlet shutoff valves, oxidizer flow control valve, tank pressurizing valves, and engine pumping and miscellaneous hardware. Emphasis is placed on fabrication and testing of the specific baseline hardware configurations as opposed to workhorse configurations. In the previous study under Contract NAS8-28989, it was established that an inactive control system will satisfy the engine control and operational requirements, and therefore, breadboard control system hardware is not necessary.

Major component testing will be initiated with rig tests of a torch igniter for the Derivative IIA engine to better define ignition operating limits of the component at OTV operation and propellant conditions. This testing is planned to be accomplished on B-9 test stand. These detailed test requirements are delineated in DVS STE 8. Other major component testing in addition to the rig testing of the torch igniter includes new turbomachinery bearing rig tests on B-13 test stand and rig testing of the fuel pump and turbine, oxidizer pump and gear drive, and oxidizer low speed inducer on B-1 and B-17 test stands. These detailed test requirements are delineated in DVS STE 4.

Control system component testing, the other major component testing for the Derivative IIA engine, is directed toward establishment of an integrated valve and control system capable of controlling the engine selectively in the tank head idle, maneuvering thrust, and full thrust modes. The control system in the tank head idle mode includes an oxidizer flow control, thrust control, and turbine bypass valve. The control system testing will consist primarily of flow bench calibrations to demonstrate accuracy, repeatability, and response time of the components. Functional and integrated systems testing will also be completed to ensure satisfactory operation during the engine test program and during engine service life. The detailed test requirements for the control system components are delineated in DVS STE 1 and 2. Additional component testing will include the injector, reoptimized for a mixture ratio of 6.0 (DVS STE 5), extendible nozzle and related hardware (DVS STE 6), plumbing and miscellaneous hardware (DVS STE 3), and a GOX heat exchanger (DVS STE 7) verification test. Tests primarily consists of bench or laboratory testing to verify as many of the component design requirements as possible before being installed on an engine assembly.


Milestones
Key Decision Points
Engine Program
Design and Fabrication
Component Test
Engine Test
GPD
Initial
PFC
FFC
AEDC
Facility Reactivation
Components
Engine
Vehicle Support
Product Support
Production
IDR
1st Comp Test
1st Turbopump Test
1st Engine Test
PDR
CDR
PFC
1st Engine Test Nozzle Extension
FFC
1st Operational Engine
Start Dev
LLT Procurements
1st Parts List
1st Ground Test Engine
PFC Parts List
FFC Parts List
75 Tests (est)
375 Tests (est)
300 Tests (est)
100 Tests (est)
B-1, B-17 Turbopump
B-9 Igniter
B-13 Bearings
Definition
E-6
E-7
Mockups
Spec, ICD, and Math Model
12
24
36
48
60
72
84
96
Months from Go-Ahead
FD 155697

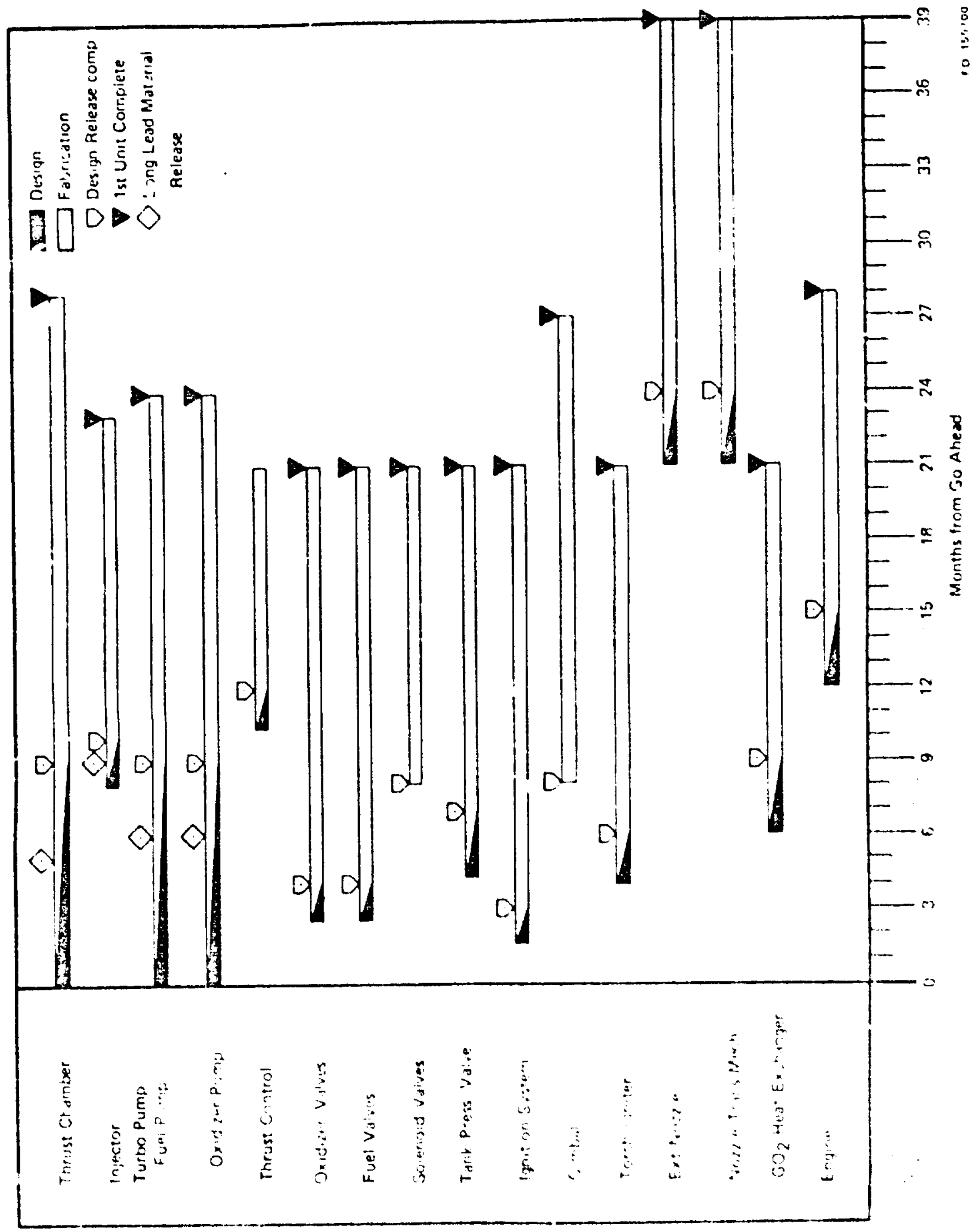


Figure 2-2. Derivative IIA Fabrication Schedule – First Development Unit

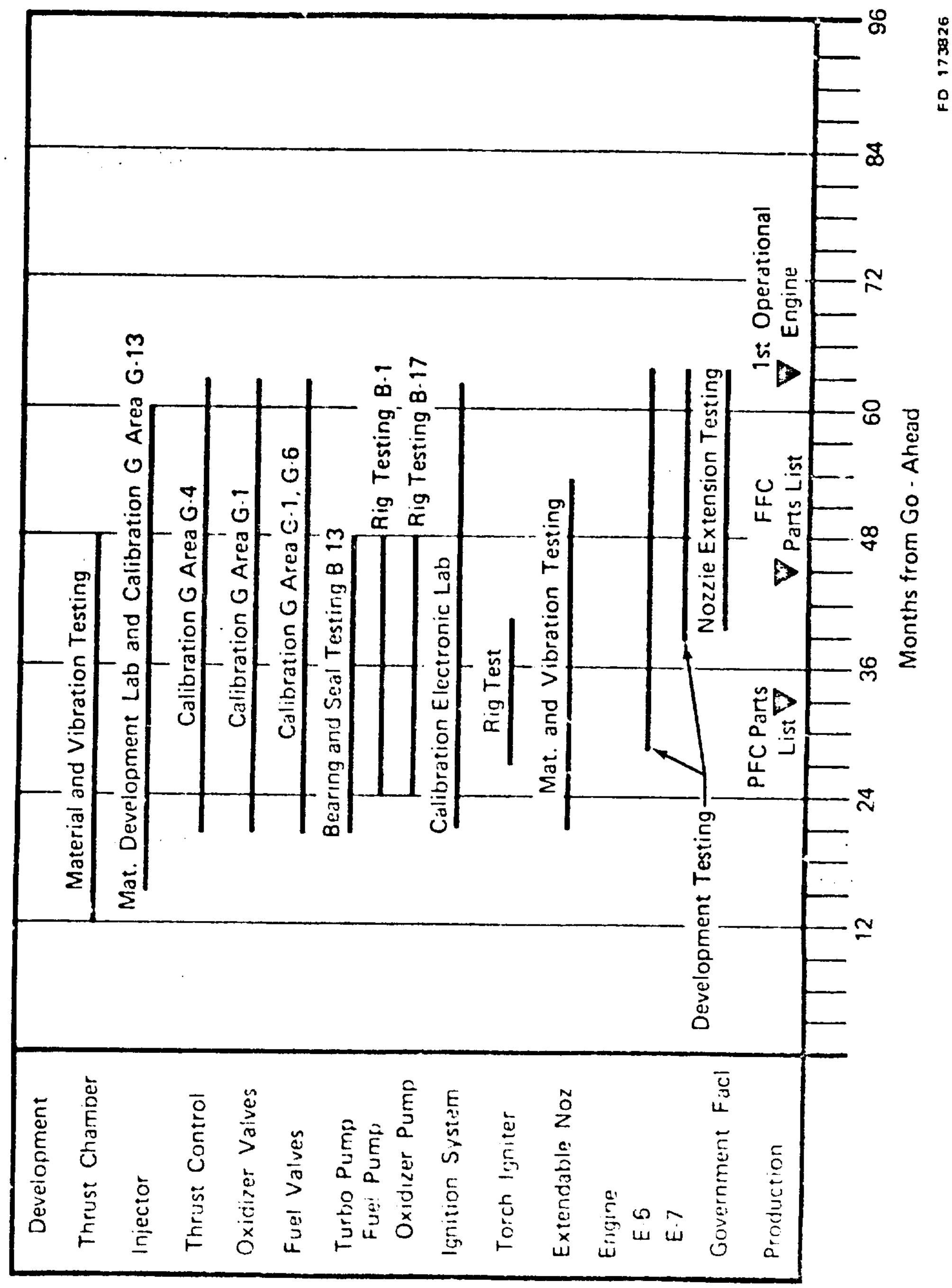


Figure 7-3. Derivative IIA Development Program Total Test Plan

Because engine tests are significant cost items in the development of rocket engines, component testing will be used to eliminate those failure modes that can be resolved at a lower hardware assembly level, thus reducing total engine system testing. Engine design and verification requirements are delineated in DVS-STE-10.

To accomplish a program efficiency, emphasis is placed on the following:

- Identification of each requirement and its verification
- Verification of requirements at the lowest hardware level practicable and as early in the program as possible
- Use of overstressing testing to accelerate failure mode detection at the subcomponent, component and engine levels.

The design verification specification documents identify each new design requirement and assumption and its verification. These documents are preliminary and contain some requirements that are to be determined. Each DVS document includes the design requirements for the components or engine, the number of hardware components required, the number and types of tests required, and the verification method. Section 7.4 of this report, Design Verification Specifications, describes DVS formulation.

A listing of the preliminary DVS documents generated during the Contract NAS 8-28989 study for the new and modified RL10A-3-3 components are listed below:

1.	DVS-STE-1	Sequenced Valves
	(1)	Fuel Inlet Shutoff Valve (FISOV)
	(2)	Oxidizer Inlet Shutoff Valve (OISOV)
	(3)	Turbine Bypass Valve (TBV)
2.	DVS-STE-2	Pressure-Operated Valves
	(1)	O[illegible]r Tank Pressurization Valve (OTPV)
	(2)	[illegible]ank Pressurization Valve (FTPV)
	(3)	[illegible]ontrol Valve (GOV)
	(4)	[illegible]er Flow Control (OFC)
3.	DVS-STE-3	Flight Instrumentation
4.	DVS-STE-4	Fuel Pump and Turbine
	(1)	Oxidizer Pump and Drive System
	(2)	Oxidizer Low Speed Inducer
5.	DVS-STE-5	Thrust Chamber Assembly
6.	DVS-STE-6	Extendible Nozzle
	(1)	Extendible Nozzle Translating Mechanism
7.	DVS-STE-7	GOX Heat Exchanger
8.	DVS-STE-8	Torch Igniter
9.	DVS-STE-9	Plumbing and Miscellaneous Hardware
10.	DVS-STE-10	Engine

The DVS's establish the program requirements in terms of numbers of hardware verifying a single design without redesign and reverification iterations. From the preliminary component and engine DVS design and verification requirements, about ten equivalent engine sets of hardware, 58 engine builds (including rebuilds), and 550 engine tests were determined necessary to accomplish the baseline Derivative IIA verification program objectives through Final Flight Certification. It was estimated that about 49 months would be necessary to accomplish the baseline Derivative IIA engine development DVS program component and engine design, fabrication, assembly and test verification requirements. The Verification Program Schedule is shown in Figure 7-4 and the control task required to accomplish the schedule is shown in Table 7-1. Redesign and reverification effort must be allowed to arrive at a realistic estimate of the total development program requirements. Historical RL10 design, fabrication and test experience was used as the basis for establishing a total baseline Derivative IIA engine development effort. The DVS effort was then subtracted from the baseline effort to yield the allowance for redesign and verification, which is about 40% of the total development effort. This appears to be reasonable based upon the difficulty of the development program and the amount of directly related experience and is illustrated in Figure 7-5.

It was estimated that about 850 engine tests over a period of 34 months, combined with a 30-month design support, fabrication, and initial component tests period, would be necessary to accomplish the total baseline Derivative IIA engine development program objectives. Duration of the overall development effort is estimated at 64 months. These total program requirements are based on previous RL10 engine modification history and similar concept history. Development of the RL10A-3-3 engine model required about 1000 engine tests during a 24-month test period and a 33-month overall development program duration.

Five active engines were selected for the engine development program based on the above considerations and particular characteristics of the expander-turbine power cycle. A total of about 90 engine builds, including rebuilds, are projected for the Derivative IIA engine total development program as compared with 175 for the equivalent RL10A-3-3 development program. About 26 equivalent engine sets of hardware are needed to support the total assembly and test programs.

Fabrication and testing of the Derivative IIA engine can be accomplished in existing RL10A-3-3 facilities. To accomplish the engine test program, two vertical test stands, E-6 and E-7, will be used. test Stand E-6 is now used for acceptance testing of the operational RL10A-3-3 engines being delivered to the NASA-LeRC for Centaur launch vehicles and will be used in this program for testing Derivative II engines with a primary nozzle only, i.e., without a nozzle extension. Test stand E-7, now inactive, will be reactivated for tank head idle thrust, maneuvering thrust, and full thrust level testing of engines with a truncated nozzle extension. The major stand special test equipment that is planned for installation in E-7 test stand, shown in Figure 7-6 is required to provide an accurate simulation of predicted propellant conditions under the zero gravity conditions encountered in space.

The high-area-ratio nozzle engine testing full nozzle extension can be accomplished at some test facility other than P&WA, such as the Arnold Engineering and Development Center (AEDC) test stand J-3. For this development program, the AEDC J-3 test stand is considered as the baseline, and the program development costs reflect this approach

A summary of the test facilities estimated for the Derivative IIA engine development program is presented in Section 7.5.

Ground support equipment (GSE) development for the baseline Derivative IIA engine is described in Section 7.6 The GSE required for the baseline Derivative IIA design, maintenance and operational modes was identified. A listing of these items is contained in the Operational and Flight Support Plan, Volume III, Part 2 of P&WA Final Report FR-6011, Design Study of RL10 Derivatives.

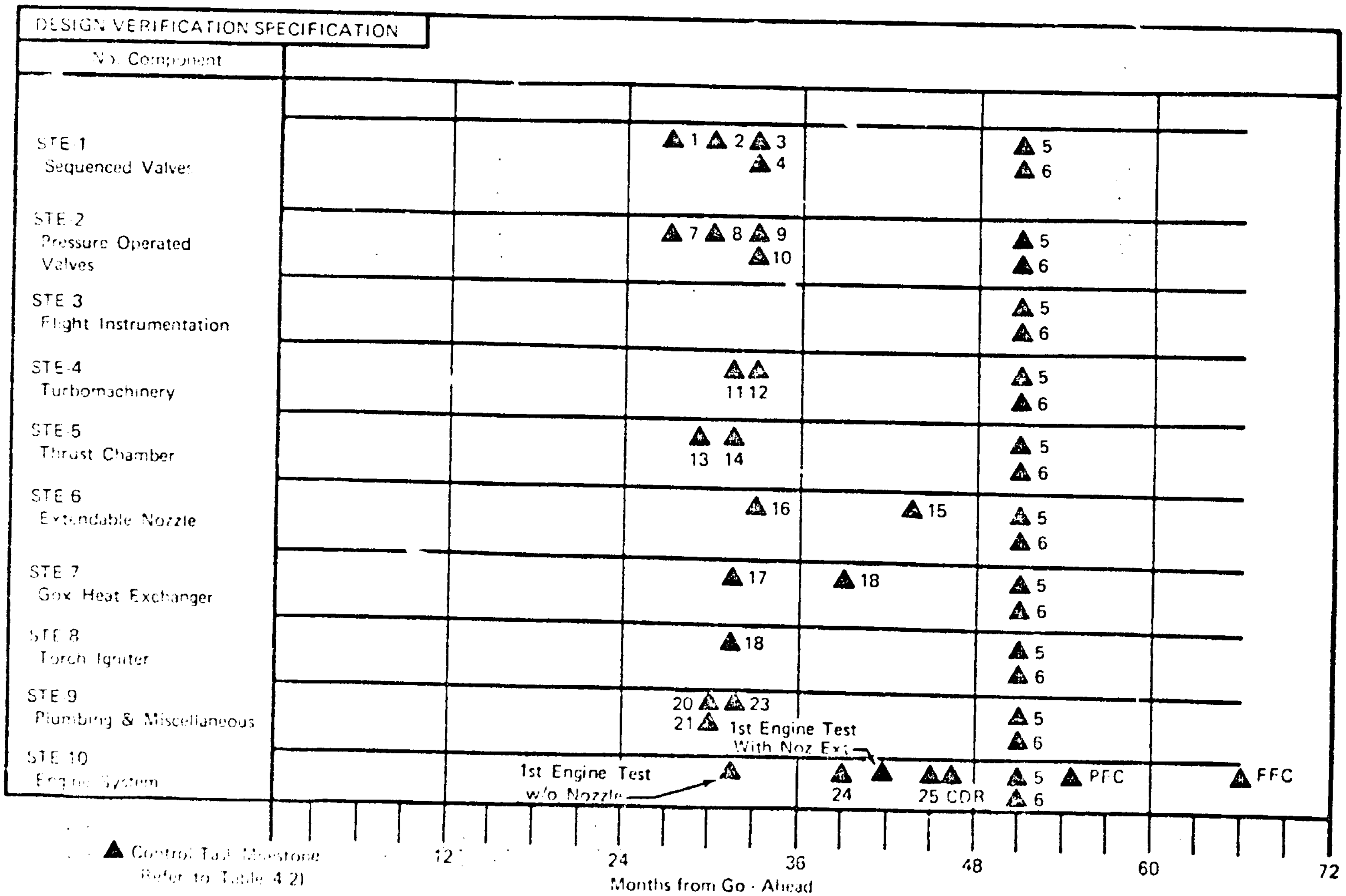


Figure 7-1. Derivative IIA Engine Design Verification Program Schedule

TABLE 7-1. CONTROL TASK DEFINITIONS

Control Task No	*Definition*
1	Establish, by conducting one waterflow test on one fuel inlet shutoff valve (FISV), oxidizer inlet shutoff valve (OISV) and turbine bypass valve (TBV), that the effective area of the component is adequate for proper operation in the engine flow system. This testing will be accomplished prior to committing the component to the first engine hot-fire test.
2	Establish, by conducting one vibration test for a minimum of TBD minutes in each axis, that the component will sustain without detrimental effects, engine static firing environment. This testing will be accomplished prior to committing the component to the first engine hot fire test.
3	Establish, by conducting endurance cycle tests on one unit for a minimum of TBD percent of the cycles specified, that the mechanical elements of the valve will function properly when subjected to engine static firing testing. This testing will be accomplished prior to committing the component to the first engine hot-fire test.
4	Establish, by conducting one functional test on one FISV, OISV and TBV that the component will meet the operational requirements of the engine. This testing will be accomplished prior to committing the FISV, OISV and TVB to the first engine hot-fire test.
5	Validate, by means of correlation to engine system test data, each requirement listed in the Requirements Source Index whose source is other than the Interface Control Document (ICD) or Contract End Item (CEI) Specification; such requirements validation shall be accomplished prior to Critical Design Review (CDR). Those requirements that cannot be validated by correlation to engine system test data are exempted from this control task. Specific requirements to be validated are coded in the Requirements Source Index.
6	Establish early detection of potential problem areas and inadequacies related to hardware design by subjecting the component or engine to all overstress testing specified in this applicable DVS, with the exception of that overstress testing planned for accomplishment at the engine system level under simulated altitude conditions. Such overstress testing, exclusive of the noted exception, will be accomplished prior to CDR.
7	Establish, by conducting one waterflow test on one oxidizer tank pressurizing valve (OTPV), fuel tank pressurizing valve (FTPV), GO_2 valve (GOV) and oxidizer flow control (OFC) that the effective area of the component is adequate for proper operation in the engine flow system. This testing will be accomplished prior to committing the component to the first engine hot-fire test.
8	Establish, by conducting one vibration test for a minimum of TBD minutes in each axis, that the component will sustain without detrimental effects, engine static firing environment. This testing will be accomplished prior to committing the component to the first engine hot fire test.
9	Establish, by conducting endurance cycle tests on one unit for a minimum of TBD percent of the cycles specified, that the mechanical elements of the valve will function properly when subjected to engine static firing testing. This testing will be accomplished prior to committing the component to the first engine hot-fire test.
10	Establish, by conducting one functional test on one OTPV, FTPV, GOV and OFC that the component will meet the operational requirements of the engine. This testing will be accomplished prior to commiting the component to the first engine hot fire test.
11	Establish, by conducting functional and performance tests on the fuel pump, oxidizer low-speed inducer, and main oxidizer pump, that the pumping performance of the major components in the Turbopump Assembly is adequate for proper operation of the engine fuel and oxidizer flow systems. This testing will be accomplished prior to committing the Turbopump Assembly to the first engine hot fire test.
12	Establish, by conducting endurance cycle tests on the fuel pump, oxidizer low speed inducer, and main oxidizer pump for a minimum of TBD percent of the cycles specified, that the mechanical elements of the Turbopump Assembly, including the gear drive train, will function properly when subjected to engine static firing testing. This testing will be accomplished prior to committing the Turbopump Assembly to the first engine hot fire test.
13	Establish, by conducting a minimum of one series of individual chamber tube [illegible] flow tests, that the chamber coolant flow is adequate to provide sufficient cooling to the thrust chamber [illegible]

TABLE 7-1. CONTROL TASK DEFINITIONS (Continued)

Control Task No	*Definition*
14	Establish, by conducting a minimum of one series of injector cold flow tests, that the mixture ratio characteristics are adequate to provide the required mixing efficiency (C* ≥98.1%). This testing will be accomplished prior to comitting the thrust chamber assembly to hot-fire tests.
15	Establish by conducting a minimum of one series of individual nozzle panel and extendible nozzle assembly cold flow tests, that the coolant flow is adequate to the extendible nozzle. This testing will be accomplished prior to committing the extendible nozzle to hot-fire tests.
16	Establish early detection of potential problem areas and inadequacies related to hardware design by subjecting the specified extendible nozzle components to overstress testing specified in this DVS. Such overstress testing will be accomplished prior to engine hot-fire tests.
17	Establish, by a minimum of (TBD) tests at Design Point on an oxygen heat exchanger, that the oxygen temperature rise and hydrogen temperture drop are within an average value of ±(TBD)°R of nominal and oxygen and hydrogen pressure losses are within ±(TBD) psid of the Engine Cycle Sheets. In addition, the oxygen heat exchanger must demonstrate stable operation. This testing will be accomplished prior to commi ting the oxygen heat oxidizer to engine testing.
18	Establish, by a minimum of (TBD) engine tests at tank head idle, pumped idle and full thrust, that the oxygen temperature rise and hydrogen temperature drop are within ±(TBD)°R of nominal and oxygen and hydrogen pressure losses are within ±(TBD) psid of the Engine Cycle Sheets over the range of engine operating conditions, as in TBD. In addition, the oxygen heat exchanger must demonstrate stable operation.
19	Establish, by a minimum of two 10-sec tests at Design Point on a torch igniter, that the hot gas temperature is within an average value of ±(TBD) of nominal and coolant flow requirements are within ±(TBD) lb/sec of the Engine Cycle Sheets. This testing will be accomplished prior to committing the torch igniter to engine testing.
20	Establish, by conducting inspections of fluid lines (as listed in Tables I, II and III of DVS-STE-9) that each line is in compliance with all of the requirements defined by the applicable drawings and process specifications. These inspections will be accomplished prior to committing any line to the first engine hot-fire test.
21	Establish, by demonstrating the satisfactory completion of the proof pressure tests, as specified for the line (as listed in Tables I, II, and III of DVS-STE-9), that these lines are structurally acceptable. This testing will be accomplished prior to committing any line to the first engine hot-fire test.
22	Establish, by conducting inspections of plumbing brackets and static seals, that each part conforms to the applicable drawing requirements and process specifications. These inspections will be accomplished prior to committing a part to the first static engine hot fire test.
23	Establish, by conducting one vibration test for a minimum of (TBD) minute in each axis, that the major fluid lines (TBD) will sustain without detrimental effects, the engine static firing environment. This testing will be accomplished prior to committing the line to the first engine hot fire test.
24	Establish, through the conduct of 10 hot fire tests on one engine system (truncated nozzle extension), that the engine is capable of safe start and shutdown operation at Tank Head Idle (THI), Maneuver Thrust (MT) and at Mainstage (MS) between mixture ratios of 5.5 and 6.5. This testing will be accomplished prior to committing the engine to testing with full nozzle extension.
25	Establish, through the conduct of 20 hot fire tests on one or more engine systems (truncated nozzle extension) that the engine is capable of safe start throughout the range of engine inlet conditions [illegible] capable of mainstage operation over the thrust range from THI Level to MS Level and mixture [illegible] from [illegible] to [illegible] at MS Level, mixture ratio [illegible] at MT Level and mixture ratio 4.0 at THI Level. In addition, safe operation with a full nozzle extension at MT and MS [illegible] established. This testing will [illegible] accomplished prior to the release of long lead [illegible] production engines.
26	Establish, through the conduct of [illegible] hot fire tests [illegible] nozzle extension [illegible] engine is capable of THI, MT and MS operation [illegible] from [illegible] at MS [illegible] the full range of inlet conditions. This testing [illegible] start of final assembly [illegible] first flight engine.

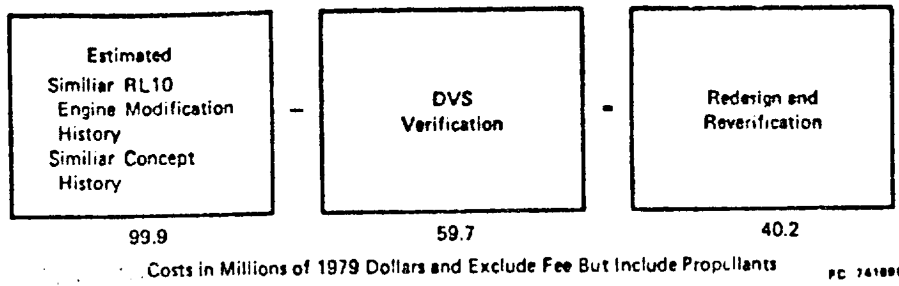


Figure 7-5. Derivative IIA Engine DDT&E Program Estimating, Approach and Results

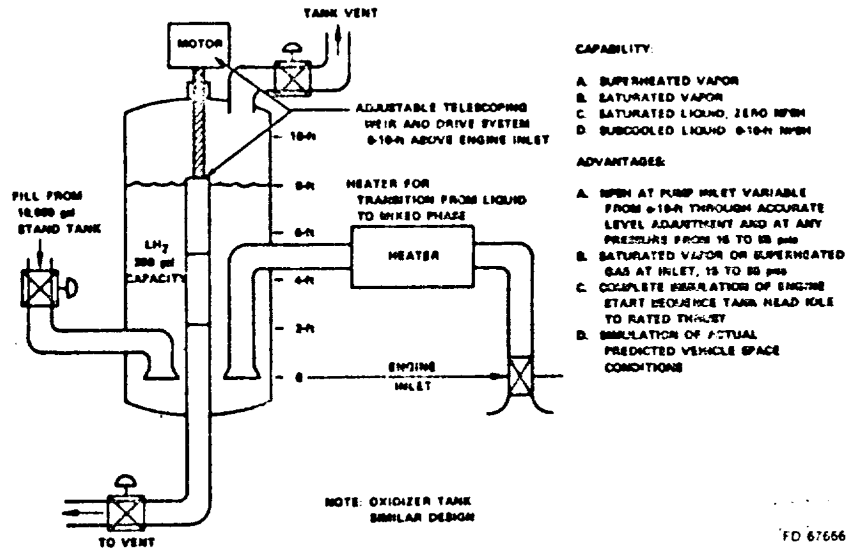


Figure 7-6. Proposed E-7 Test Stand Modification to Provide Accurate Simulation of Predicted Propellant Conditions Under Zero Gravity Conditions

The estimated propellant and ancillary fluid requirements for the baseline Derivative IIA engine total development program are included in the program cost estimate.

The estimated quantity requirements are:

LH_2	7,000,000 lb
LO_2	30,000,000 lb
LN_2	13,000,000 lb
GN_2	280,000,000 scf
GH_2	2,400,000 scf

7.2.3 Derivative IIB Engine

The baseline Derivative IIB engine design includes the same features as the Derivative IIA engine design except for the inlet condition requirements. Whereas the Derivative IIA engine features a "zero" net positive suction head (NPSH) operation capability (< 40% vapor for both fuel and oxidizer) at steady-state full thrust operation, the Derivative IIB engine requires a positive NPSH (> 14 ft NPSH fuel and > 7.5 ft NPSH oxidizer). This relaxation in inlet condition requirements allows the use of the RL10A-3-3 engine "as-is" parts list turbomachinery. Thus, the resultant programmatic effect is a shorter development program duration (59 months) that features 100 less engine tests and three less equivalent engine sets of hardware through FFC. The Derivative IIB engine consists of a RL10A-3-3 engine with:

1. Recontoured, high-expansion-ratio, two-position nozzle
2. Reoptimized injector
3. Tank head idle mode capability
4. Pumped idle mode capability (25% full thrust, herein designated maneuvering thrust)
5. Autogenous pressurization (fuel and oxidizer).

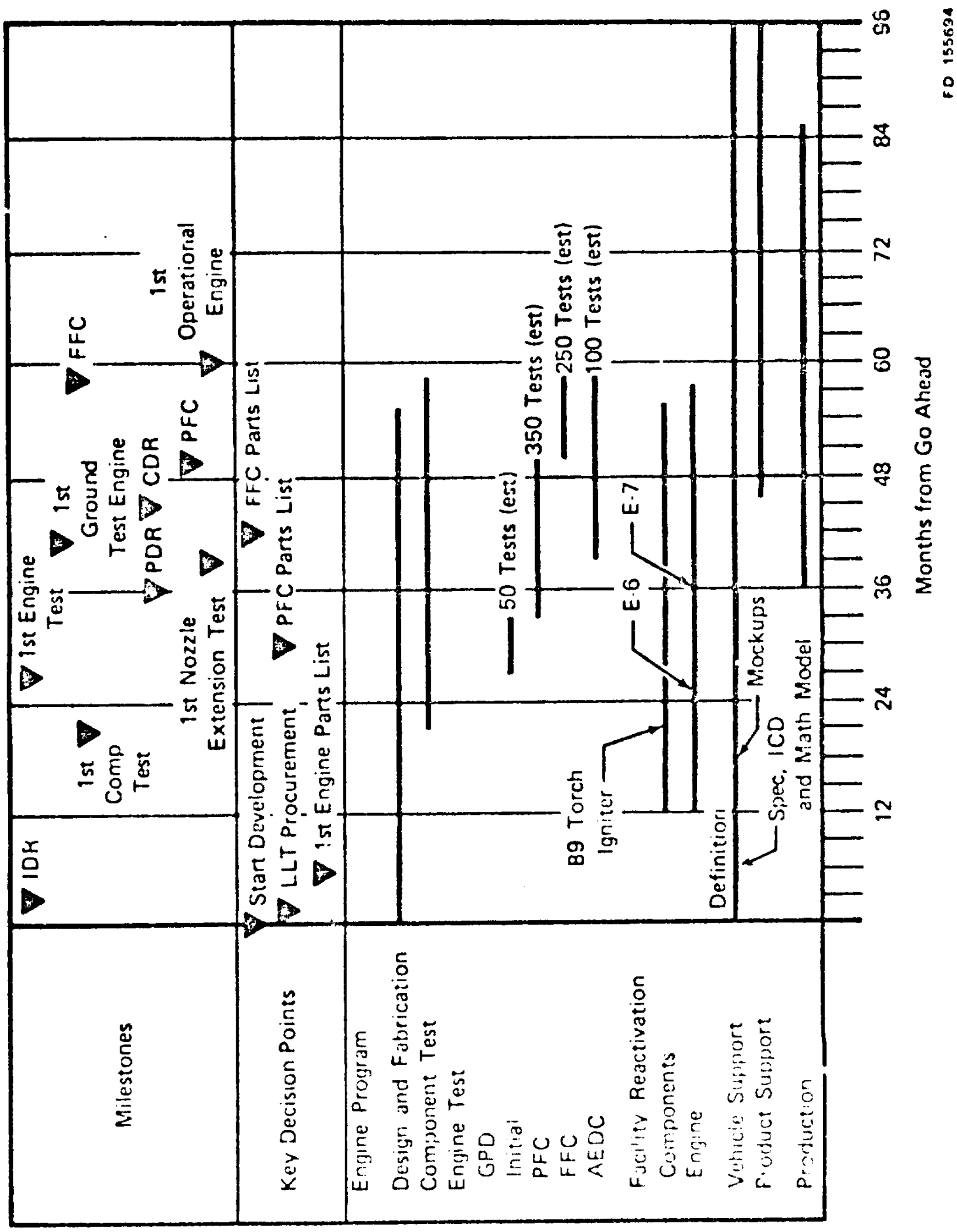
The development program for the baseline Derivative IIB engine will require about 59 months of design, fabrication, and test effort. This effort will encompass three design, build, test cycles to FFC (Initial, PFC and FFC configurations). Figure 7-7 depicts the development schedule presenting the major program milestones and key decision points as well as the total engine development program. The design and fabrication schedules for this program are shown in Figure 7-8, and the program test plan is shown in Figure 7-9.

The design and fabrication and test effort will be carried out in the same manner as described for the baseline Derivative II engine, under Section 3, Subsection 2.1, except the turbomachinery and bearing design, build and verification testing, and the propellant inlet valve design, build and verification testing is not required since these components are the same component configuration as those already qualified and used in the operational RL10A-3-3 engine used in the Centaur vehicle launch program.

Preliminary DVS documents generated during the previous study for the new and modified RL10A-3-3 components, which require design verification for the baseline Derivative IIB engine are essentially the same as those required for the Derivative IIA engine (refer to Table 7-2) with the following exceptions:

1. The DVS STE-1 requirements for sequenced valves in the case of the Derivative IIB engine development are applicable to the Turbine Bypass Valve only since the fuel and oxidizer inlet shutoff valves are the same configuration as the qualified RL10A-3-3 engine valves.
2. The DVS STE-4 requirements for the oxidizer pump and drive system and the oxidizer low speed inducer are deleted entirely for the same reason as given for DVS STE-1 above, i.e., the operational RL10A-3-3 engine parts lists turbomachinery is used "as is" in the Derivative IIB engine design.

The DVS's establish the program requirements in terms of numbers of hardware and tests for testing levels estimated for verification of the component and engine design at PFC and FFC. The requirements specified in these documents are based on verifying a single design with no redesign iterations. From the preliminary component and engine DVS design and verification requirements, about nine equivalent engine sets of hardware ([illegible] engine builds) (including rebuilds) and 500 engine tests were determined to be necessary to accomplish the baseline Derivative IIB verification program objectives through Final Flight Certification. The overall baseline Derivative IIB component/engine development effort was estimated using the same approach as for the Derivative IIA. The estimate for redesign and reverification is about 40% of the total development effort. The approach and results are shown in Figure 7-10. It was estimated that about 48 months would be necessary to accomplish the baseline Derivative IIB engine development DVS program component and engine design, fabrication, assembly and test verification requirements.


Milestones
IDR
1st Comp Test
1st Engine Test
1st Nozzle Extension Test
PDR
1st Ground Test Engine
CDR
PFC
FFC
1st Operational Engine
Key Decision Points
Start Development
LLT Procurement
1st Engine Parts List
PFC Parts List
FFC Parts List
Engine Program
Design and Fabrication
Component Test
Engine Test
GPD
Initial
PFC
FFC
AEDC
50 Tests (est)
350 Tests (est)
250 Tests (est)
100 Tests (est)
Facility Reactivation
Components
Engine
B9 Torch Igniter
E-6
E-7
Vehicle Support
Product Support
Production
Definition
Spec, ICD and Math Model
Mockups
12
24
36
48
60
72
84
96
Months from Go Ahead
FD 155694

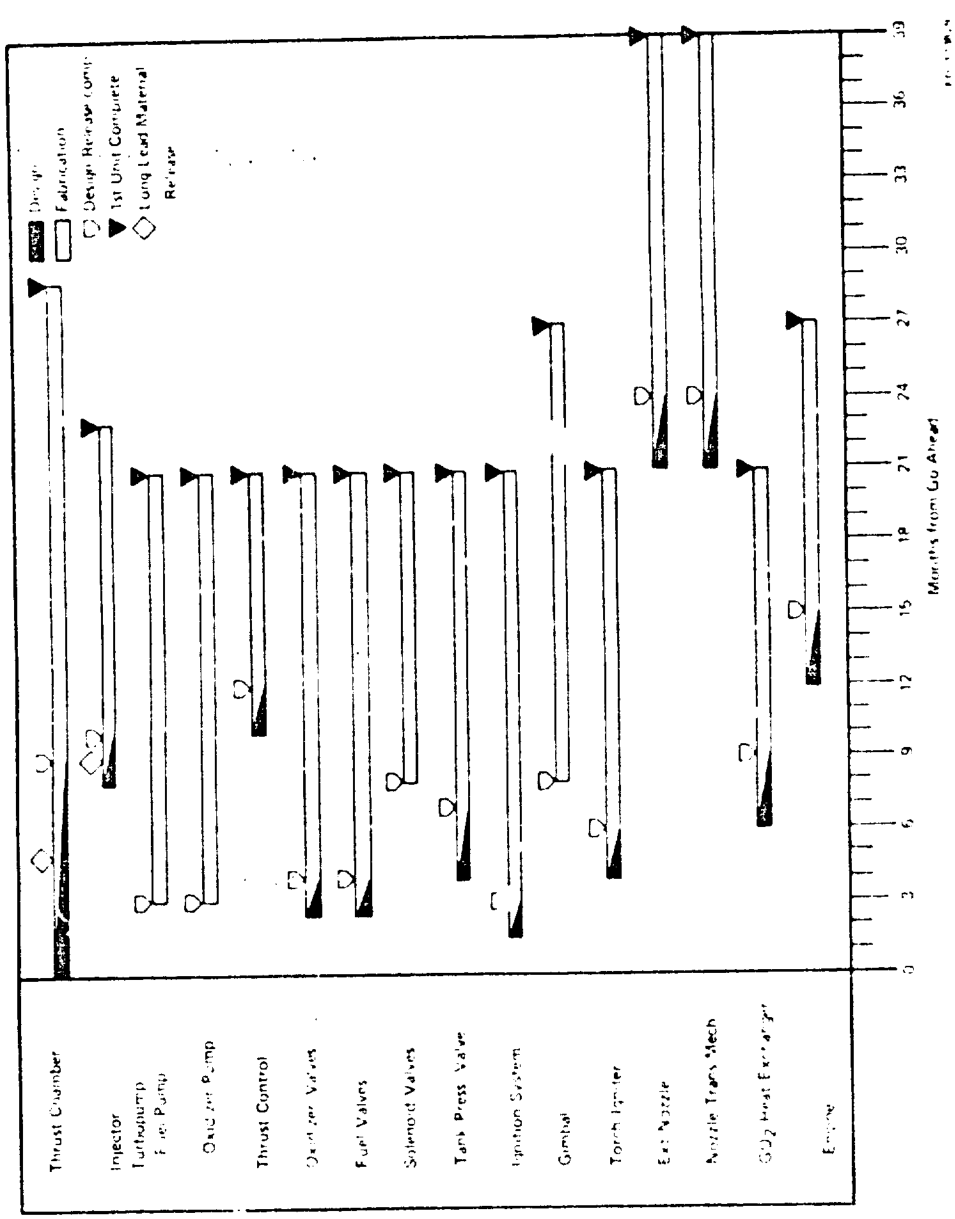


Figure 7-8. Derivative HB Fabrication Schedule, First Development Unit

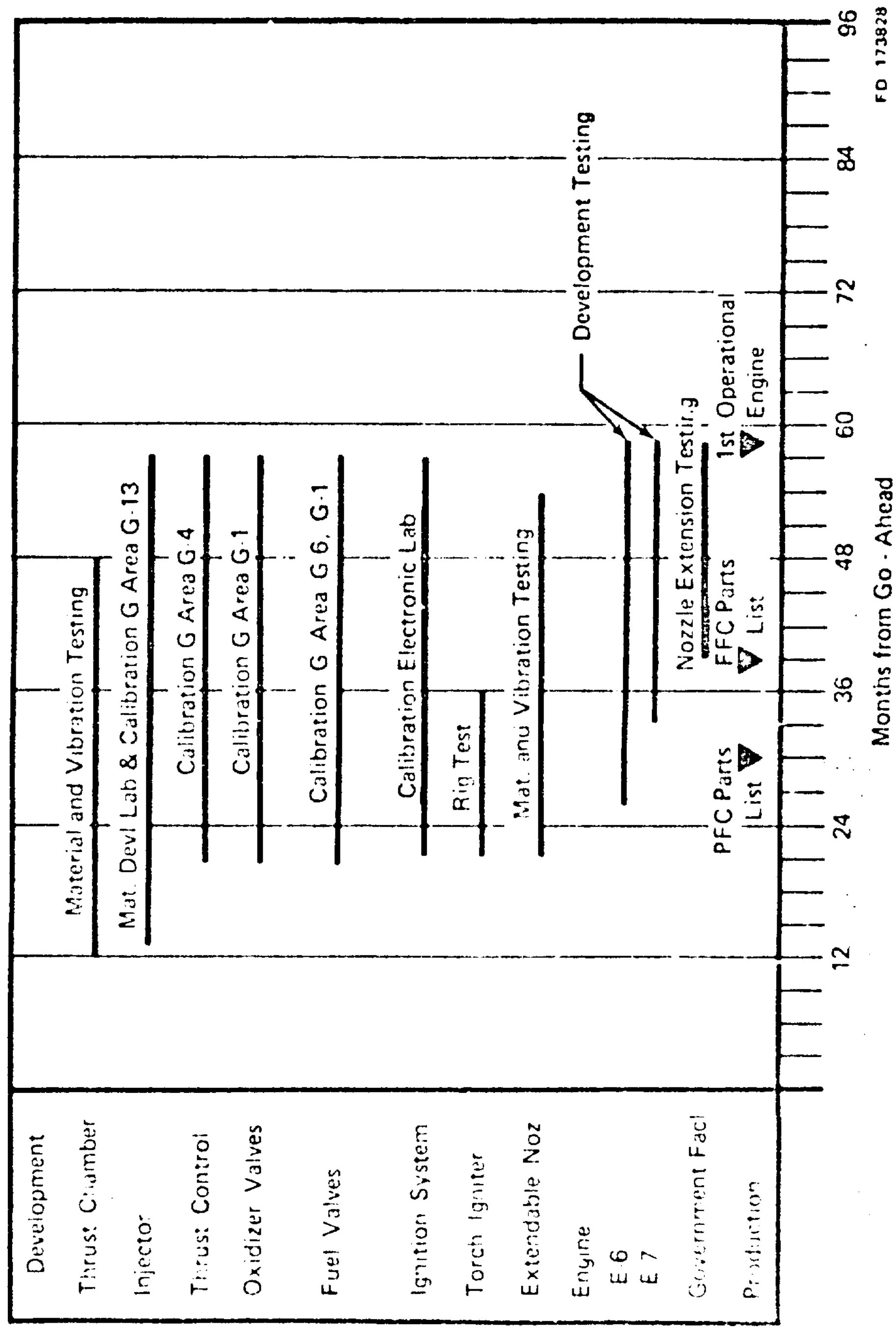
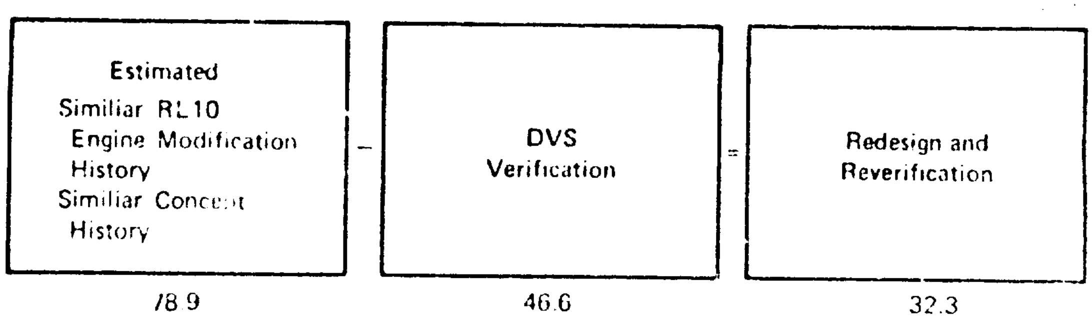


Figure 7-9. Derivative IIB — Development Program Total Test Plan

TABLE 7-2. RL10 DERIVATIVE ENGINE DVS LIST

Component(s)	*DVS STF No*	*Candidate Engine DVS Requirement* *IIA*	*IIB*
Sequence Valves	1		
Fuel Inlet Shutoff Valve		X	
Oxidizer Inlet Shutoff Valve		X	
Turbine Bypass Valve		X	X
Pressure Operated Valves	2		
Oxidizer Tank Pressurization Valve		X	X
Fuel Tank Pressurization Valve		X	X
GOX Flow Control Valve		X	X
Oxidizer Flow Control		X	X
Flight Instrumentation	3	X	X
Fuel Pump and Turbine	4	X	
Oxidizer Pump and LSI Drive		X	
Oxidizer Low-Speed Inducer		X	
Thrust Chamber Assembly	5	X	X
Extendible Nozzle	6	X	X
Extendible Nozzle Translating Mechanism		X	X
GOX Heat Exchanger	7	X	X
Torch Igniter	8	X	X
Plumbing and Miscellaneous Hardware	9	X	X
Engine	10	X	X



Costs in Millions of 1979 Dollars and Exclude Fee But Include Propellants

Figure 7-10 Derivative IIB Engine DDT&E Program Estimating Approach and Results

It was estimated that about 750 engine tests over a period of 32 months, combined witl 27-month design support, fabrication, and initial component test period, would be necessary accomplish the total baseline Derivative IIB engine total development program objectiv Duration of the overall development effort is estimated at 59 months. These total progr requirements are based on previous RL10 engine modification history and similar conce history, and current material lead time. Development of the RL10A-3-3 engine model requir about 1000 engine tests during the 24-month test period and a 33-month overall developme program duration.

Five active engines were selected for the Derivative IIB engine development program bas on the above considerations and particular characteristics of the expander turbine power cyc A total of about 80 engines, including rebuilds, will be used for the Derivative IIB engine te development program as compared with 175 for the equivalent RL10A-3-3 development progr About 23 equivalent engine sets of hardware are planned to support the total assembly and t programs.

Fabrication and testing of the Derivative IIB engine will be accomplished in the existi RL10A-3-3 facilities. To accomplish the engine test program, two vertical test stands, E-6 ar E-7, will be used. Test stand E-6 is now used for acceptance testing of the operational RL10A-3 engines being delivered to the NASA LeRC for Centaur and will be used in this program t testing Derivative IIB engine with a primary nozzle, i.e., without a nozzle extension. Test stan E-7, now inactive, will be reactivated for the tank head idle thrust, pumped idle thrust, and fu thrust level engine testing of engines with a truncated nozzle extension. The major stand speci test equipment that is planned for installation in E-7 test stand, previously shown in Figure 1- is required to provide an accurate simulation of predicted propellant conditions under the ze gravity conditions encountered in space.

Because the Derivative IIB engine uses the RL10A-3-3 engine parts list turbomachinery component test facility and test benches B-1, B-17, and B-13, reactivation and/or modificatio is not required. All other facilities listed in Subsection 7.5 are required for the Derivative III engine development program. A summary of the test facilities estimated for the Derivative III engine development program is presented in Section 7.5.

The high-area-ratio nozzle engine testing can be accomplished in the Arnold Engineerin and Development Center (AEDC) test stand J-3. It is necessary to make modifications to thes stands to make them operational for this testing. For this development program the AEDC J- test stand is considered the baseline, and the program development costs reflect this approac

Ground support equipment (GSE) development for the baseline Derivative IIB engine i described in Section 7.6. The GSE required for the baseline Derivative IIB design, main tenance and operational modes was established. A listing of these items for the Derivative III engine, which are identical with the Derivative IIA engine, is contained in the Operational anc Flight Support Plan, Volume III, Part 2 of the previous study.

The estimated propellant and ancillary fluid requirements for the baseline Derivative III engine total development program are included in the program cost estimate.

The estimated requirements are

LH_2	5,000,000 lb
LO_2	[illegible] lb
LN_2	[illegible] lb
GN_2	[illegible] scf
GH_2	[illegible] scf

7.2.4 Derivative IIC Engine

The third candidate baseline engine is the RL10 Derivative IIC, which is a modification of the basic RL10A-3-3 engine by the addition of a high area-ratio, two-position nozzle and a recontoured primary nozzle to provide increased performance for an early operational OTV.

The development program for the baseline Derivative IIC engine design consists of 37 months of design, fabrication and test effort. This effort will encompass one design-build-test cycle to FFC, i.e., only the Final Flight Configuration will be developed. The development schedule showing the major program milestones, key decision points, and the total engine development program is shown in Figure 7-11. The baseline Derivative IIC test plan is shown in Figure 7-12.

Major component testing will be initiated with tank pressurization valve testing followed by torch igniter testing.

Engine testing will be accomplished in the same manner as described in the baseline Derivative IIA engine program development and will begin 31 months after start of development.

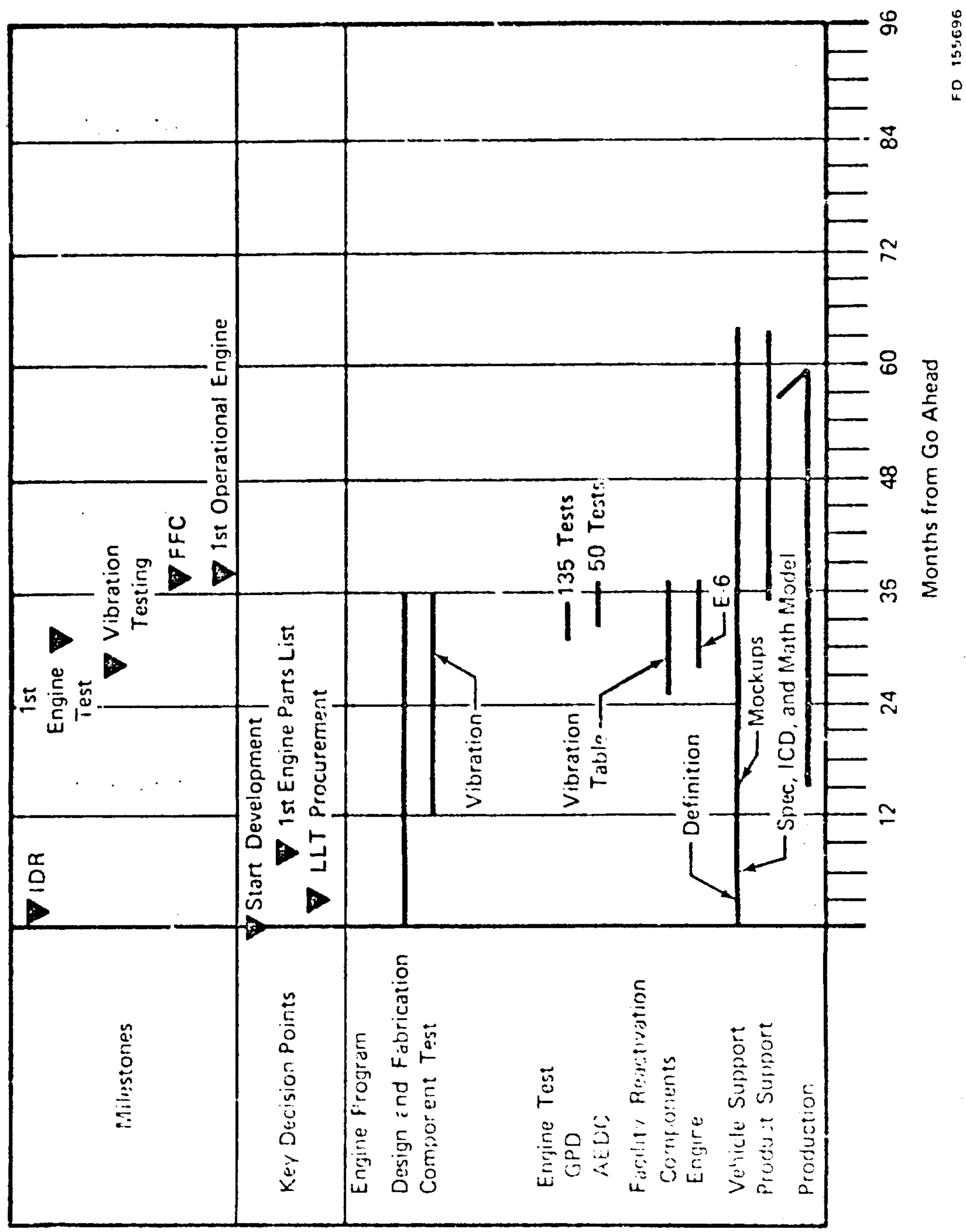
A preliminary listing of the Design Verification Specification documents required for the design verification of the new and modified RL10A-3-3 components used in the Derivative IIC engine is given below:

1. Pressure Operated Valves
   - Fuel Tank Pressurization Valve
   - Oxidizer Flow Control
2. Flight Instrumentation
3. Thrust Chamber Assembly
4. Extendible Nozzle
5. Plumbing and Miscellaneous Hardware
6. Derivative IIC Engine

Although DVS's were not formulated as a part of the Derivative IIC engine development program plan, an estimate of the DVS program requirements was made. This was made from DVS requirements comparison of the Derivative IIC engine configuration with the engineering judgment to adjust the Derivative IIB engine DVS requirements to a level comparable to Derivative IIC engine DVS requirements. The resulting DVS program requirements that were determined to be necessary for the Derivative IIC engine are about [illegible] equivalent sets of engine hardware, engine builds (including rebuilds) and [illegible] engine system tests.

It was estimated that about [illegible] months would be necessary to accomplish the baseline Derivative IIC engine DVS program component and engine design verification, assembly and test verification [illegible].

Four active engines or a total of 10 engines [illegible] development [illegible]. [illegible] and assembly and test programs.


Milestones
IDR
1st Engine Test
Vibration Testing
FFC
1st Operational Engine
Key Decision Points
Start Development
1st Engine Parts List
LLT Procurement
Engine Program
Design and Fabrication
Component Test
Vibration
Engine Test
GPD
AEDC
135 Tests
50 Tests
Facility Reactivation
Components
Engine
Vibration Table
E-6
Vehicle Support
Product Support
Production
Definition
Mockups
Spec, ICD, and Math Model
12
24
36
48
60
72
84
96
Months from Go Ahead
FD 155696

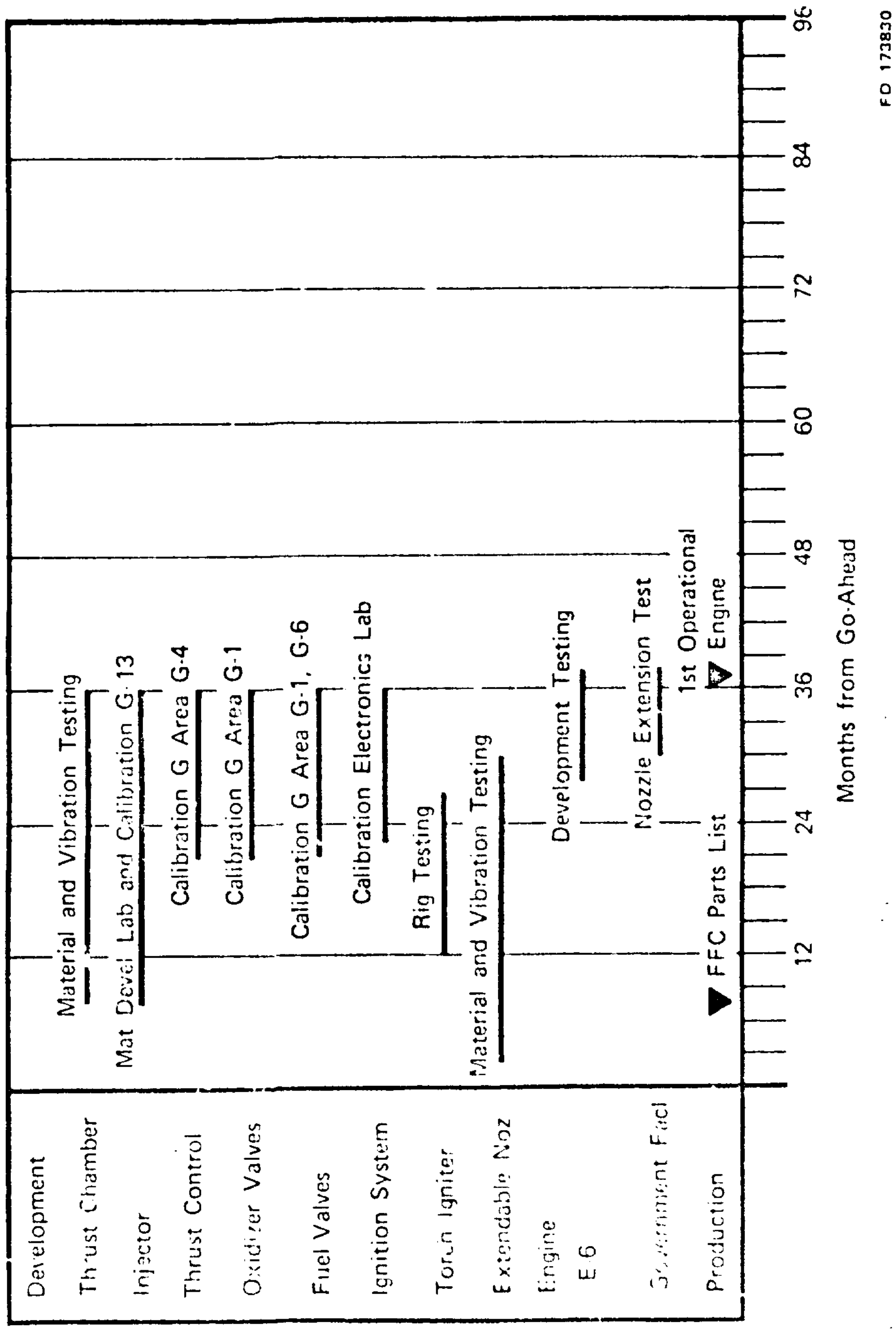


Figure 7-12. Derivative IIC Development Program Total Test Plan

Historical RL10 design, fabrication and test experience formed the basis for estimating the duration of the overall baseline Derivative IIC development and the number of engine tests required. It was estimated that about 185 engine tests over a period of 6 months, combined with a 31-month design and fabrication period, will be necessary to accomplish the baseline Derivative IIC engine development program objectives. Duration of the overall development effort is estimated at 37 months.

Special test equipment, test stand reactivation, and modifications to accomplish the baseline Derivative IIC engine program requirements are much less than those described for the baseline Category IIB engine programs, since the only major difference from the basic RL10A-3-3 engine is the addition of the two-position nozzle. The characteristics of this nozzle require that engine testing with the nozzle installed be accomplished in a test facility such as the J-3 stand at AEDC.

Ground support equipment requirements will consider modifications to existing GSE designs. New GSE will be provided as necessary and developed concurrently with engine development. The GSE development is described in Section 7.6.

The estimated propellant and ancillary fluid requirements for the baseline Derivative IIC engine total program quantity requirements are estimated to be:

LH_2	1,200,000 lb
LO_2	5,200,000 lb
LN_2	2,300,000 lb
GN_2	49,000,000 scf
GH_e	410,000 scf

7.2.5 Category IV Engine

The fourth candidate baseline engine is the RL10 Category IV. This new engine, specifically designed for use in the OTV, is based on the RL10 and uses RL10 engine design concepts where possible. It is interchangeable with the RL10 Derivative IIA engine and has the same operating modes, but is lighter, shorter, has a higher performance and a longer life.

The development program for the baseline Category IV engine design consists of 80 months of design fabrication and test effort. This effort encompasses three design build-test cycles to FFC; i.e., initial, Preliminary Flight Configuration (PFC) and Final Flight Configuration (FFC). The development schedule showing the major program milestones, key decision points, and the total engine development program is shown in Figure 7-13. The baseline Category IV test plan is shown in Figure 7-14.

Major component testing will be initiated with high-pressure fuel turbopump and oxidizer turbopump and bearing testing followed by low-pressure fuel and oxidizer pump, valve and control system testing.

Engine testing will be accomplished in the same manner as described in the baseline Derivative IIA engine program development and will begin 39 months after start of development.

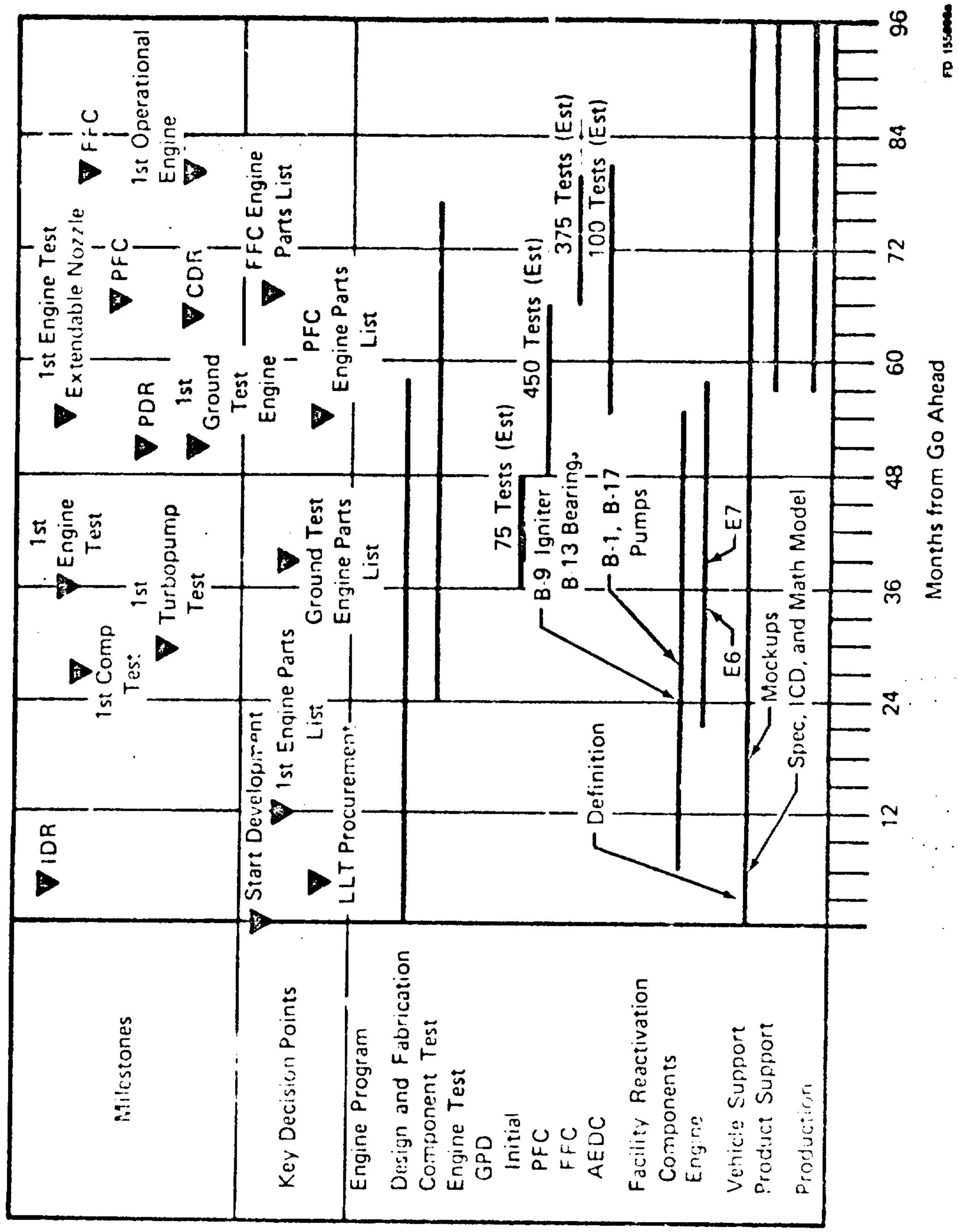


Figure 7-13. Category IV Program Schedule and Major Program Milestones

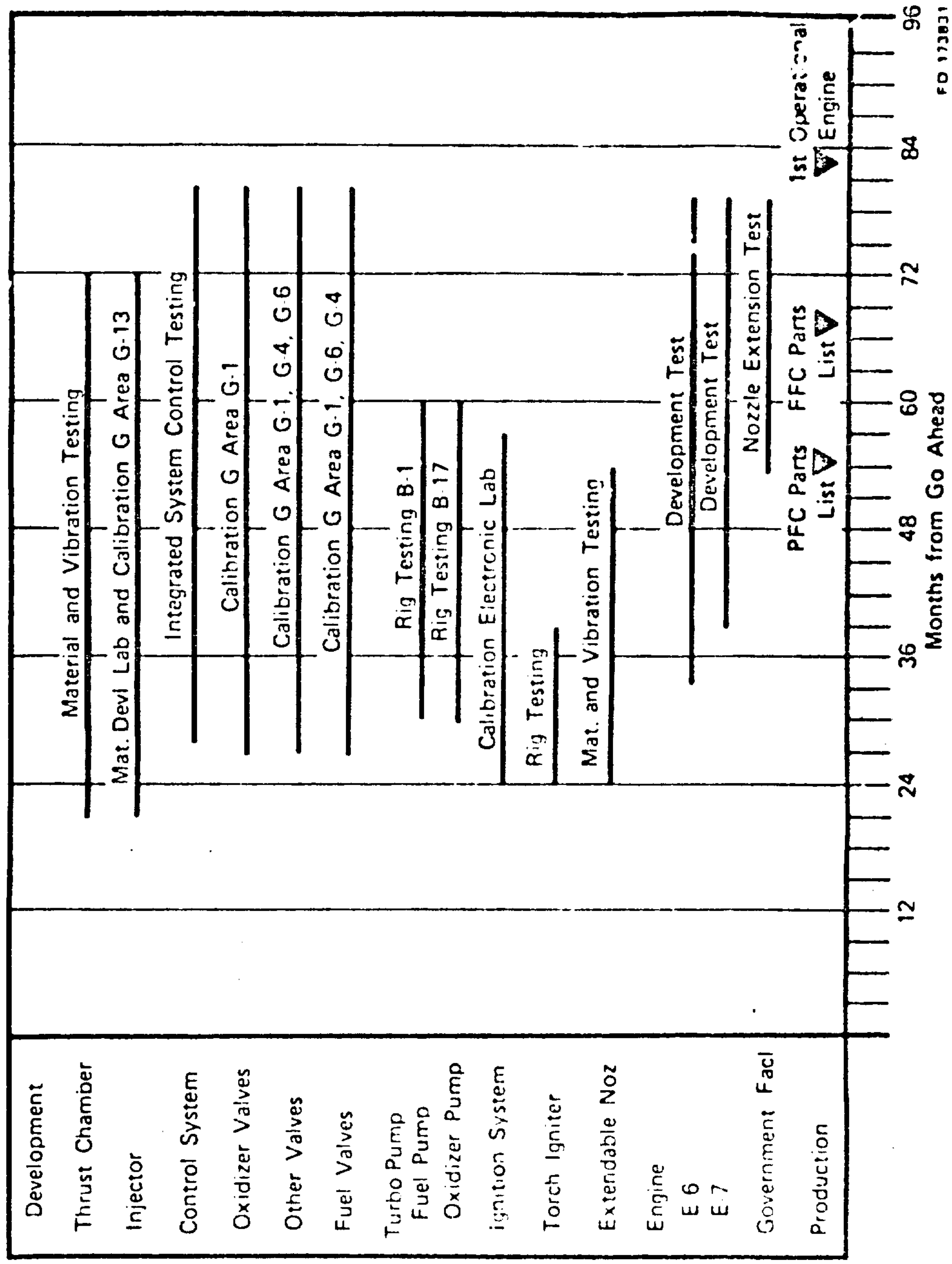


Figure 7-11. Category IV Development Program Total Test Plan

A preliminary listing of the Design Verification specification documents required for the design verification of the Category IV engine is given below:

1. Sequenced Valves
   - Fuel Inlet Shutoff Valve
   - Oxidizer Inlet Shutoff Valve
2. Main Fuel Control
3. Pressure-Operated Valve
   - Oxidizer Tank Pressurization Valve
   - Fuel Tank Pressurization Valve
   - GOX Control Valve
   - Oxidizer Flow Control
4. Flight Instrumentation
5. Turbopumps
   - Fuel Low-Speed Inducer
   - High-Pressure Fuel Turbopump
   - Oxidizer Low-Speed Inducer
   - High-Pressure Oxidizer Turbopump
6. Thrust Chamber Assembly
7. Extendible Nozzle
   - Extendible Nozzle Translating Mechanism
8. GOX Heat Exchanger
9. Torch Igniter
10. Plumbing and Miscellaneous Hardware
11. Category IV Engine.

Although DVS's were not formulated as a part of the Category IV engine development program plan, an estimate of the DVS program requirements was made. This was made from DVS requirements comparison of the Category IV engine configuration with the engineering judgment to upgrade the Derivative IIA engine DVS requirements to a level comparable to Category IV engine DVS requirements. The resulting DVS program requirements necessary for the Category IV engine are: 11 equivalent sets of engine hardware, 84 engine builds (including rebuilds) and 600 engine system tests.

It was estimated that about 69 months would be necessary to accomplish the baseline Category IV engine DVS program component and engine design, fabrication, assembly, and test verification requirements. The approach for estimating the elements of the engine development program costs and the results are shown in Figure 7-15.

Six active engines (for a total of 130 engines, including rebuilds) were selected for the total development program. About 30 equivalent engine sets of hardware are needed to support the total assembly and test programs.

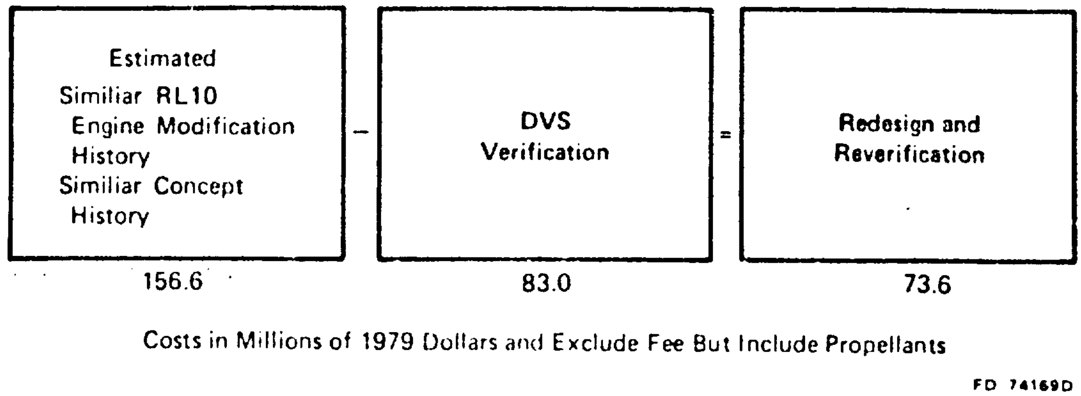


Figure 7-15 Category IV Engine DDT&E Program Estimating Approach and Costs

Historical RL10 design, fabrication and test experience formed the basis for estimating the duration of the overall baseline Category IV development and the number of engine tests required. It was estimated that about 1000 engine tests over a period of 44 months, combined with a 36-month design, fabrication and initial component test period, will be necessary to accomplish the baseline Category IV engine development program objectives. Duration of the overall development effort is estimated at 80 months. RL10 engine development to the first RL10 Preliminary Qualification required about 1200 engine tests and a 64-month development program.

Special test equipment, test stand reactivation, and modifications to accomplish the baseline Category IV engine program were assumed to be the same as described for the baseline Category IIA engine program.

Ground support equipment requirements will consider modifications to existing GSE designs. New GSE will be provided as necessary and developed concurrently with engine development. The GSE development is described in Section 7.6. A listing of these items of GSE for the Category IV engine is contained in the Operational and Flight Support Plan, Volume III, Part 2 of the final report of the previous study.

The propellant and ancillary fluid requirements for the baseline Category IV engine total development program are estimated to be:

LH_2	9,000,000 lb
LO_2	29,000,000 lb
LN_2	17,000,000 lb
GN_2	365,000,000 scf
GH_2	3,000,000 scf

7.2.6 Advanced Expander Cycle Engine

The fifth candidate baseline engine is the Advanced Expander Cycle. This new engine, specifically designed for use in the OTV, is based on the ground rules taken from the contract Statement of Work (shown in Table 7-31).

TABLE 7-3. OTV ENGINE REQUIREMENTS

The requirements for the OTV engine applicable to a vehicle of the type envisioned have been derived from numerous NASA in-house and contracted vehicle and systems studies and are:

1. The engine will operate on liquid hydrogen and liquid oxygen propellants.

2. Engine design and materials technology are to be based on 1980 state of the art.

3. The engine must be capable of accommodating programmed and/or commanded variations in mixture ratio over an operating range of 6:1 to 7:1 during a given mission. The effects on engine operation and lifetime must be predictable over the operating mixture ratio range.

4. The propellant inlet temperatures shall be 162.7°R for the oxygen boost pump and 37.8°R for the hydrogen boost pump. The boost pump inlet NPSH at full thrust shall be 2 ft for the oxygen pump and 15 ft for the hydrogen pump.

5. The service free life of the engine cannot be less than 60 start/shutdown cycles or 2 hr accumulated run time, and the service life between overhauls cannot be less than 300 start/shutdown cycles or 10 hr accumulated run time. The engine shall have provisions for ease of access, minimum maintenance, and economical overhaul.

6. The engine, when operating within the nominal prescribed range of thrust, mixture ratio, and propellant inlet conditions shall not incur during its service life chamber pressure oscillation, disturbances, or random spikes greater than ±5% of the mean steady-state chamber pressure.

7. The engine nozzle is to be a contoured bell with an extendible/retractable section.

8. Engine gimbal requirements are +15 deg and −6 deg in the pitch plane and ±6 deg in the yaw plane.

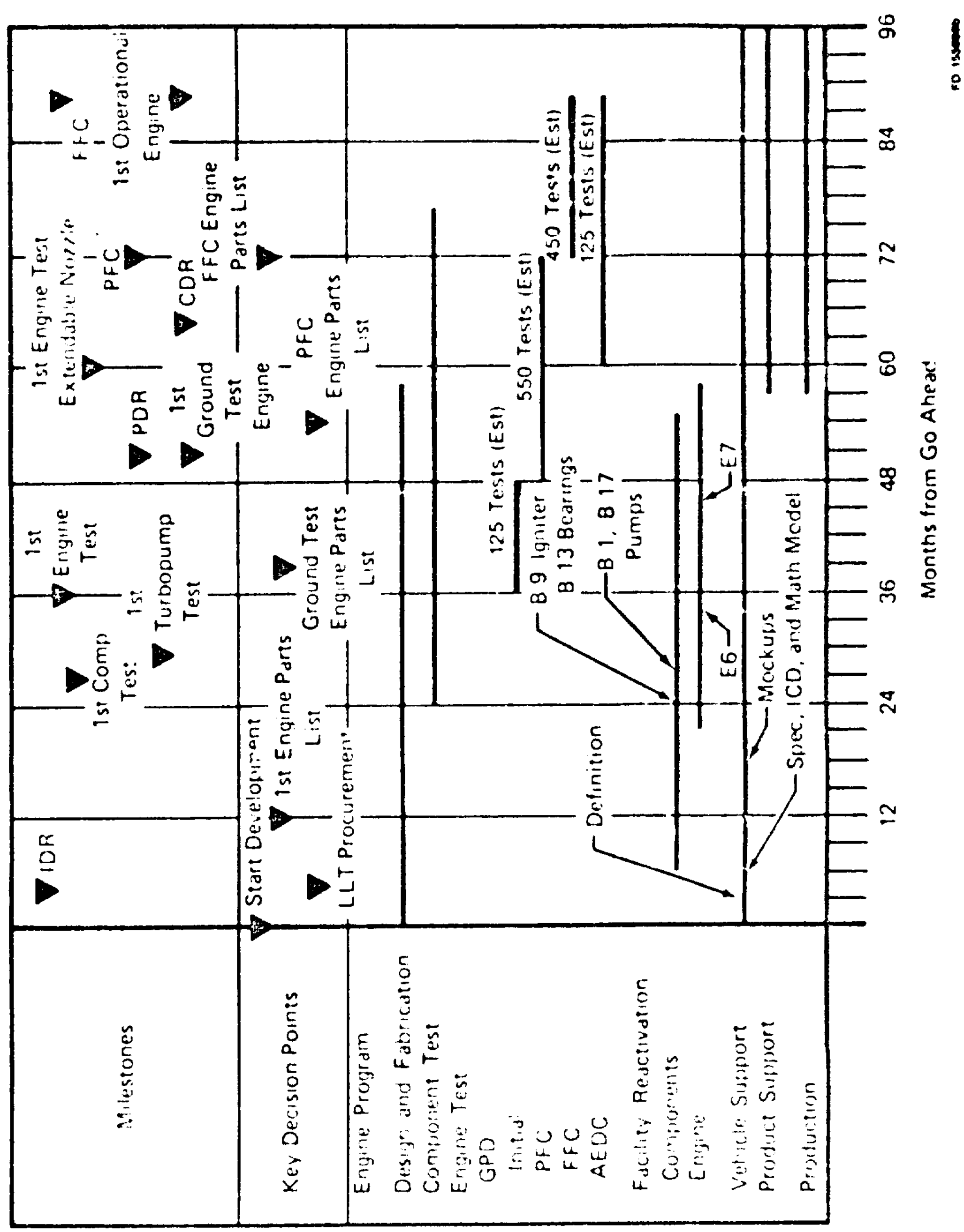
9. The engine is to provide gaseous hydrogen and oxygen autogenous pressurization for the propellant tanks.

10. The engine is to be man rated.

The development program for the baseline Advanced Expander Cycle engine design consists of 86 mo of design, fabrication and test effort. This effort encompasses three design-build-test cycles to FFC; i.e., initial, Preliminary Flight Configuration (PFC) and Final Flight Configuration (FFC). The development schedule showing the major program milestones, key decision points, and the total engine development program is shown in Figure 7-16. The baseline Advanced Expander Cycle engine design and fabrication schedules are shown in Figure 7-17.

Major component testing will be initiated with high-pressure fuel turbopump and oxidizer turbopump and bearing testing followed by low-pressure fuel and oxidizer pump, valve and control system testing.

Engine testing will be accomplished in the same manner as described in the baseline Derivative IIA engine program development and will begin 35 mo after start of development.



[illegible] Engine Development Schedule and Major Program Milestones

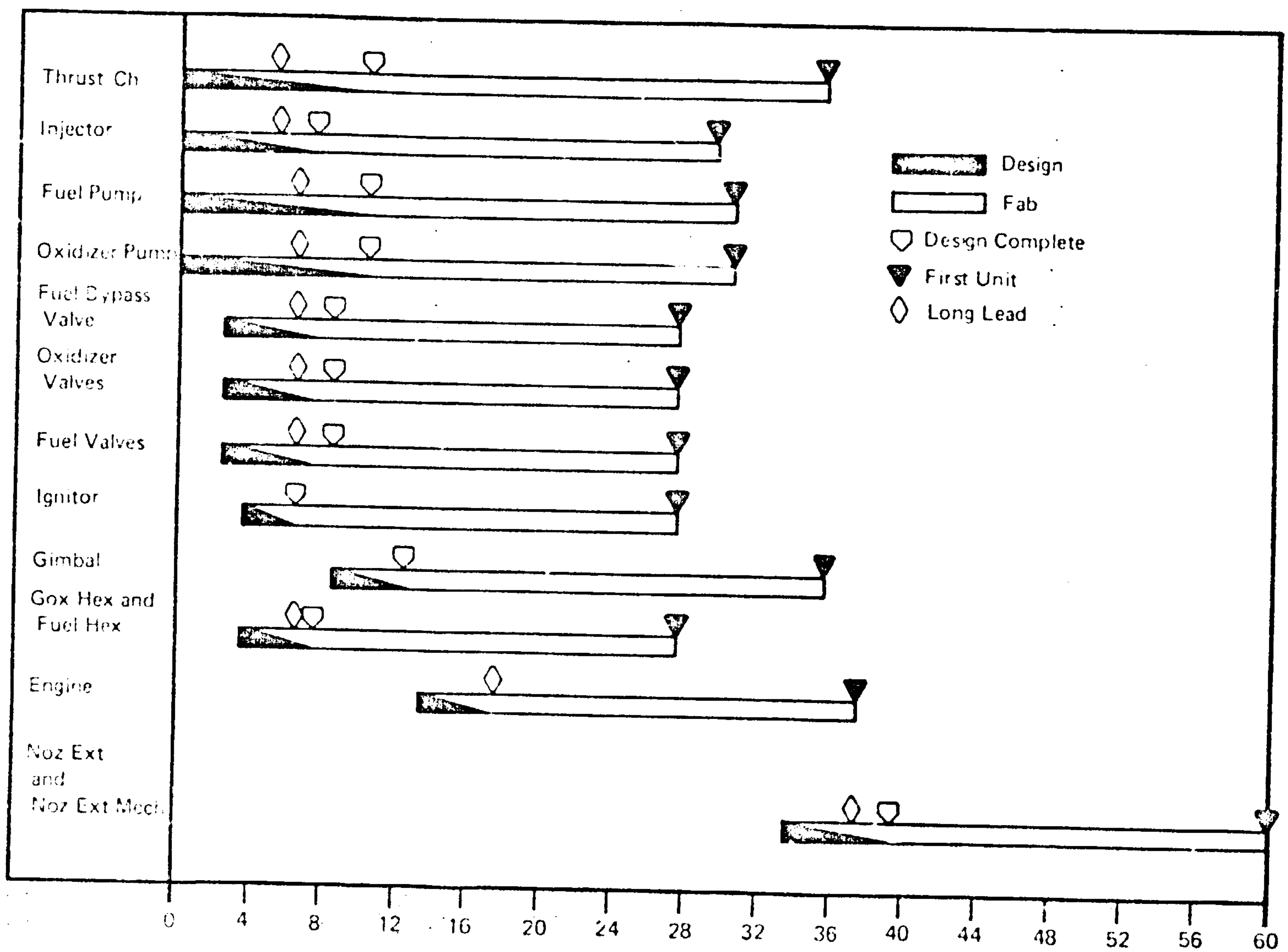


Figure 7.17 Advanced Expander Cycle Engine Fabrication Schedule, First Development Unit

A preliminary listing of the Design Verification specification documents required for the design verification of the Advanced Expander Cycle engine is given below:

1. Sequenced Valves
   - Fuel Inlet Shutoff Valve
   - Oxidizer Inlet Shutoff Valve
2. Main Fuel Control
3. Pressure-Operated Valves
   - Oxidizer Tank Pressurization Valve
   - Fuel Tank Pressurization Valve
   - GOX Control Valve
   - Oxidizer Flow Control
4. Flight Instrumentation
5. Turbopumps
   - Fuel Low Speed Inducer
   - High Pressure Fuel Turbopump
   - Oxidizer Low Speed Inducer
   - High Pressure Oxidizer Turbopump
6. Thrust Chamber Assembly
7. Extendible Nozzle
   - Extendible Nozzle Translating Mechanism
8. GOX Heat Exchanger
9. Torch Igniter
10. Plumbing and Miscellaneous Hardware
11. Advanced Expander Cycle Engine

Although DVS's were not formulated as a part of the engine development program plan, an estimate of the DVS program requirements was made. This was made from DVS requirements comparison of the Advanced Expander Cycle engine configuration with the engineering judgment to upgrade the Derivative IIA engine DVS requirements to a level comparable to Advanced Expander Cycle engine DVS requirements. The resulting DVS program requirements necessary for the Advanced Expander Cycle engine are 15 equivalent sets of engine hardware, 108 engine builds (including rebuilds) and 750 engine system tests. Eight active engines (for a total of 180 engine builds and rebuilds) were selected for the total development program. About 40 equivalent engine sets of hardware are needed to support the total assembly and test programs.

It was estimated that about [illegible] mo would be necessary to accomplish the baseline Advanced Expander Cycle engine DVS program component and engine design, fabrication, assembly, and test verification requirements.

Historical RL10 design, fabrication and test experience formed the basis for estimating the duration of the overall baseline Advanced Expander Cycle development and the number of engine tests required. It was estimated that about 1250 engine tests over a period of 54 mo, combined with a 36 mo design, fabrication and initial component test period, will be necessary to accomplish the baseline Advanced Expander Cycle engine development program objectives.

Duration of the overall development effort is estimated at 89 mo. RL10 engine development to the first RL10 Preliminary Qualification required about 1200 engine tests and a 64-mo development program.

Special test equipment, test stand reactivation, and modifications to accomplish the baseline Advanced Expander Cycle engine program were assumed to be the same as described for the baseline Category IIA engine program.

Ground support equipment requirements will consider modifications to existing GSE designs. New GSE will be provided as necessary and developed concurrently with engine development. The GSE development is described in Section 6. A listing of these items of GSE for the Category IV engine is contained in the Operational and Flight Support Plan, Volume II, Part 2 of the final report of the previous study, and are similar to those items required by the Advanced Expander Cycle Engine.

The propellant and ancillary fluid requirements for the baseline Advanced Expander Cycle engine total development program are estimated to be:

LH_2	11,250,000 lb
LO_2	49,000,000 lb
LN_2	22,000,000 lb
GN_2	450,000,000 scf
GH_2	4,000,000 scf

7.3 DESIGN VERIFICATION SPECIFICATIONS

7.3.1 General

Design Verification Specifications (DVS's) applicable to the Derivative IIA and IIB engine programs form the foundation upon which the Engine Development Program Plans can be based. The DVS's specify the design requirements and the requirements for verifying the design. Routine component and engine feasibility testing and hardware requirements, engine support, and other miscellaneous testing and hardware requirements, in addition to the DVS requirements, are not included in the individual DVS's.

7.3.2 DVS Development

Fundamental performance criteria, contained in an Engine CEI Specification, Interface Control Document or Analytical Analysis provide the design requirements to develop the engine and ensure meeting the mission requirements. Engine CEI Specification have not been formulated for the derivative RL10 engines, so use is made of the fundamental performance criteria contained in the Preliminary Interface Control Document, Baseline Tug Definition Document and other applicable documents, such as the RL10A-3-3 Model Specification No. 2265A. Since the DVS's used in this study are preliminary in nature because the program is in the study phase, component and engine design thus far is conceptual, hence an engine model specification is not available. The DVS's contain the preliminary data necessary to establish the performance and design requirements for the engine and each major component. Where data are not available, the requirements were TBD based on the assumption that as requirements become known, the TBD's would be incorporated. DVS's further describe the method of verification, verification tests and hardware requirements, analytical correlation of detail requirements to engine performance and operating requirements, and interrelationships between PFC and FFC requirements. The DVS provides references to supporting or governing documents as required, and substantiates design assumptions, test methods, and analytical data.

During the previous study effort, preliminary DVS's were formulated for the Derivative IIA and IIB engine configurations. A listing of the DVS's that are anticipated for the Derivative IIC, Category IV and Advanced Expander Cycle engines is provided in Section 7.3.4, 7.3.5, and 7.3.6, respectively.

Ten individual preliminary DVS's were prepared for the engine component such as thrust chamber assembly, sequenced and pressure-operated valves, fuel pump, etc., and for the engine. For a list of these DVS's refer back to Table 7-2.

Total development planning effort for the RL10 Derivative engines assumed that redesign and reverification is required. The 40% redesign effort estimate for the Derivative IIA and IIB engines was based on the difference of the development experience of previous RL10 engine development programs and the estimate of the DVS programs. In an on-going engine development program, the formulation of reverification test programs required for verification of component/engine redesigns is based on an assessment of the original design and verification. Reverification test programs would require both P&WA and Government approval for authority to proceed.

Control of the hardware configuration used during verification is the responsibility of the applicable DVS Project Engineer, and is monitored by the Configuration Management Organization.

DVS's initiated by a Project/Design Engineering Team are revised through a Configuration Management System when the preliminary implication is removed from the DVS's. This provides a common system for DVS's Engineering Change Notices (ECN), Engineering Releases (ER), Engine Model Specification Change Notices (SCN), and Interface Revision Notices (IRN), and also provides the internal interface to keep the DVS current with the total requirements. The method used to generate, control, and update these specifications is illustrated in Figure 7-18.

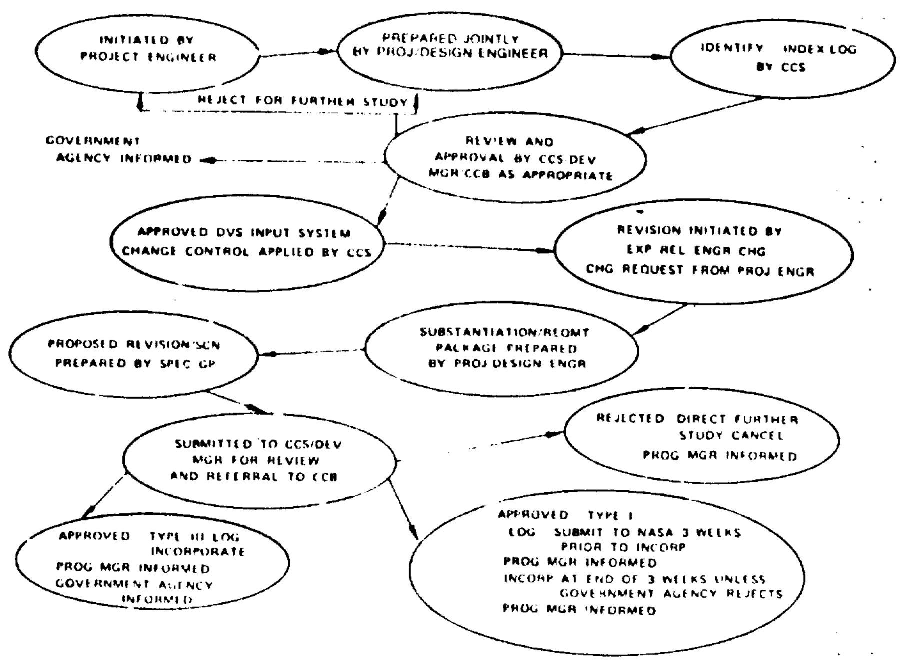


Figure 7-18 DVS System Control

7.3.3 DVS Construction

7.3.3.1 *Section I. Scope*

This section sets the bounds of applicability of the DVS. It consists of purpose, lists the major sections, and provides a statement that each design requirement is keyed to a verification method through a "Cross Reference Index." It further specifies the DVS's relationship to the Program Development Plan.

7.3.3.2 *Section II. Applicable Documents*

This section is to be determined later. When completed it will contain a list of applicable documents referenced in the text of the DVS. The list is categorized as follows:

1. Specifications
2. Standards
3. Drawings
4. Handbooks and Manuals
5. Other.

7.3.3.3 *Section III. Requirements*

The requirements section consists of the following five subsections:

1. Performance — This subsection consists of functional characteristics both primary and secondary, operability in terms of maintainability, failsafe design, useful life, environmental and safety criteria

2. CEI Definition — This subsection consists of interface requirements

3. Design and Construction — This subsection consists of general design features, selection and review of specifications and standards, materials and processes, interchangeability and replaceability, manufacturing process, electromagnetic, interface/susceptibility and identification and marking requirements

4. Reliability Assurance — This subsection states that the component/engine shall be subject to design review, FMECA and test and malfunction analysis requirements

5. Quality Assurance — This subsection states that components/engine shall be designed such that material, fabrication processes, cleanliness controls and tests may be inspected, and/or verified in accordance with Quality Assurance requirements.

7.3.3.4 *Section IV. Verification of Requirements*

This section specifies the test criteria required to verify the design requirements presented in Section III of the DVS. It is divided into the following seven major subsections:

1. Verification Test and Evaluation — This consists of verification program description and control, control task definitions and verification complete documentation

2. Analysis -- This section specifies the design requirements verification by analysis

3. Hardware Inspection -- This section specifies the design and construction and quality assurance requirements verification by inspection

4. Laboratory or Bench Tests -- This section specifies the secondary performance and operational characteristics and operability requirements verification by laboratory or bench testing

5. Subsystems Hot-Fire Tests -- This section specifies the primary performance and operational characteristics, secondary performance and operational characteristics, operability, interface and general design feature requirements by subsystem hot-fire tests

6. Engine Hot-Fire Tests -- This section specifies the primary performance and operational characteristics, secondary performance and operational characteristics, operability, interface and general design features requirements by engine hot-fire tests

7. Flight Certification -- This section specifies the certifications demonstration and flight certification documentation requirements verification.

7.4 SUMMARY OF TEST FACILITIES

7.4.1 Facilities at Government Products Division (GPD)

GPD has thirteen existing test stands, an instrumentation and ignition laboratory, humidity chamber, gimbal stand, vibration tables (small and large) and test support services planned for utilization in the OTV engine development program. Five of these test stands can be used without modifications to accomplish the planned testing. The remaining eight test stands will require modification for this program. Detail part, component subassemblies, component assemblies and engines will be tested on these test stands. Additional tests are planned to be conducted by Rocket Assembly, Quality Assurance, and Materials Development Laboratory.

The function and capability of each test stand is summarized in Table 7-3.

7.4.2 Government Facilities

NASA ground rules for the previous study (NAS8-28989) called for testing of RL10 engines with large area ratio nozzle extensions to be accomplished offsite from GPD. For planning purposes, the J-3 test cell in the Engine Test Facility of the Arnold Engineering Development Center was assumed to be available.

Test Cell J-3 is a vertical test cell in which rocket propulsion systems up to 200,000 lb thrust, or engines with a maximum nozzle exit diameter of 100 in., are tested under simulated high-altitude conditions. The test cell is capable of testing complete flightweight missile stages or spacecraft. The largest of two available test capsules (18 by 40 ft) is equipped with nitrogen cooled panels for simulating low temperature environments in addition to low pressures. Space propulsion systems up to 36 ft in length may be tested in the capsule.

TABLE 7-4. GPD TEST STAND UTILIZATION

GPD Stand No	*Description/Use*
E-6	Simulated altitude tests to verify engine operation and performance without a high area ratio extendible nozzle. Exhaust pressures of 0.3 psia, 0.4 psia, 0.6 psia, and 1.1 psia can be attained at start, tank head idle thrust, maneuvering (pumped idle) thrust and full thrust, respectively. (Needs minor reactivation and modification)
E-7	Simulated altitude tests to verify engine start operation under zero gravity conditions. Planned capability will include superheated vapor, saturated vapor, saturated liquid, zero NPSH, and subcooled liquid, 0-40 ft NPSH. Simulated tests to verify engine operation and performance with a truncated high area ratio extendible nozzle. Exhaust pressures of 0.08 psia, 0.3 psia, and 0.5 psia can be attained at start, tank head idle thrust, maneuvering (pumped idle) thrust and full thrust, respectively. (Need major reactivation and modification)
B-17	Low- and high-pressure oxidizer pump tests. (Derivative IIA and Category IV.) Not required for Derivative IIB. (Needs major reactivation)
B-1	Low- (Category IV) and high-pressure hydrogen pump tests. (Category IV and Derivative IIA.) Not required for Derivative IIB. (Needs major reactivation)
B-24	Temperature probe calibration at cryogenic temperatures (no modification)
B-13	Liquid hydrogen bearing and seal tests at engine operating temperatures and speeds. (Needs major reactivation)
B-21	Valve calibrations with water at flows to 100 gpm and pressures to 100 psi. (Needs major reactivation)
B-22	Valve cryogenic cycling and valve calibrations with water at flows to 100 gpm and pressures to 750 psi. (Needs major reactivation)
B-9	Thrust chamber torch igniter hot firing tests at engine operating conditions. (Deactivated, will need major reconstruction)
G-1	Valve, calibrations with liquid nitrogen at flows to 225 gpm and pressures to 925 psi (no modifications)
G-4	Thrust control cryogenic calibrations with gaseous nitrogen and gaseous helium at flows to 0.20 scfm, at pressures to 1800 psi and 2000 psi, respectively (no modifications)
G-6	Valve leakage calibration with gaseous nitrogen and gaseous helium at flows to 700,000 sccm, at pressures to 1800 psi and 2000 psi, respectively (no modifications)
G-13	Injector calibrations with water at flows to 225 gpm and pressures to 750 psi (no modifications)
Gimbal Stand	**Nonfiring gimbaling separately in the pitch and yaw planes; sinusoidal wave signal input to actuators at integral frequencies from 1 to 30 Hz. (Needs to be reactivated.)**
Humidity Chamber	Salt atmosphere testing of RL10 engine with controlled water, salt, and airflow, including dry bulb wet bulb temperature recorder controller. (Needs to be reactivated)
Vibration	
Small Table	Valve and component vibration testing MB Model C-210 vibration exciter at frequencies of 5 to 2000 Hz sine. Force output is 28,000 lb peak sine. (Needs to be reactivated)
Large Table	Complete engine horizontal or vertical vibration testing on MB Model C-10 vibration exciter at frequencies of 5 to 6000 Hz sine. Force output is 1750 lb peak sine.

The J-3 complex is equipped for both 3500- and 5000-psi gaseous helium storage, a 3500 psi gaseous nitrogen storage, and a 28,000-gal liquid nitrogen dewar. Storage for liquid propellants includes liquid oxygen, hydrocarbon fuels, and storable oxidizers and fuels and a diked rail spur for liquid hydrogen storage. High-capacity, 25-micron absolute filters are installed in the propellant systems to maintain particulate contamination at low levels.

An inplace flowmeter calibration system is installed in both the oxidizer and fuel systems to provide flowmeter calibrations in the proper fluid media at the proper pressures and temperatures using the test feedline configuration. The required modifications to J-3 test cell to permit testing of engines with high expansion ratio, extendible nozzles of large exit area have not been considered in this study.

7.5 GROUND SUPPORT EQUIPMENT DEVELOPMENT PROGRAM

7.5.1 General

New ground support equipment (GSE) for engine development programs must be developed concurrently with engine development to ensure compatibility between engine and GSE functions. Milestones for new GSE development should be integrated with engine development milestones when an ongoing program is identified, so that GSE will be provided for training and support of the first flight (PFC) engines to be delivered as well as for operational (FFC) engines. New GSE development must be an integral part of engine manufacturing and testing.

The GSE items were presented in the Design Study of RL10 Derivatives (Final Report, FR-6011 Operational and Flight Support Plan, Volume II, Part 2) and are identified as existing and new GSE. These items are classified into four major groups.

1. Mechanical Equipment — Wrenches, covers, kits, etc.
2. Electrical Equipment — Electrical checkout consoles
3. Inspection Equipment — Borescopes, leak detectors, etc.
4. Handling Equipment — Maintenance stands, slings, adapters, engine transporters, etc.

7.5.2 Identification of New Requirements

Design layout reviews are conducted informally during the design and formally at the completion of the design to identify any new GSE requirements. Engine GSE design coordination and review is conducted to achieve the following objectives.

7.5.2.1

Elimination of GSE requirements, where possible and practical, is a design objective. Engine components and parts that require servicing, inspection, or maintenance are designed to make use of standard tools instead of requiring special wrenches, pullers, fixtures, etc. When the engine design features which would be required to eliminate otherwise needed GSE significantly affect engine weight, manufacturing complexity or cost, trade studies are conducted to determine the best course of action.

7.5.2.2

Where elimination of GSE requirements is impossible or is shown to be impractical by a trade study, design coordination and layout reviews serve to identify the GSE requirements and assure compatibility of the GSE with the engine. Engine design layout reviews by the engineering team responsible for the development of GSE, the engineering teams responsible for the engine development, and the ILS-GSE team responsible for the use and support of GSE in the field ensure the requirements for GSE are kept as simple as possible, and compatability of GSE is ensured with operational requirements.

Following the design layout reviews, a maintenance engineering analysis is conducted on each maintenance significant engine item to identify the maintenance tasks, labor skills required and tooling needed, and to formally document each GSE requirement.

7.5.3 Screening of Available Assets

New GSE requirements having been identified, a survey of existing Government and P&WA equipment would be conducted to determine if there is any available equipment which can satisfactorily be used to meet these new GSE requirements.

During the screening process the possibility of using off-the-shelf, commercially available equipment will also be reviewed as an alternative to modifying existing equipment. The purpose of the screening process is to avoid the design and development of new equipment whenever possible, thereby reducing program costs.

7.5.4 Design or Modification

Where new GSE requirements have been identified which cannot be fulfilled by existing equipment or the modification of existing equipment, the design of new equipment is required.

With design approach selection of a particular item of GSE, and with preliminary drawings complete, a Preliminary Design Review (PDR) would be conducted with NASA.

7.5.5 Development

The construction of new GSE hardware is planned under the direction of the Development Group. This equipment is put in use during the engine development program and must be developed with the engine. Performance of the prototype GSE during the engine development program permits Engineering to evaluate equipment design and effectiveness. When development experience indicates change is required, GSE designs are modified and these modifications are incorporated in the prototype hardware for further evaluation. Where GSE is required primarily for operational use, and usage in the engine development program does not provide an adequate opportunity for evaluation, special tests are planned and executed by the Engineering Group.

New GSE items requiring significant development effort would be developed to meet the specific operational requirements determined and verified by the Integrated Logistics Support groups, to provide checkout of the engine systems and components, and to provide ready handling and transportation of the engine.

New GSE items, such as handling devices, slings, maintenance stands, helium and nitrogen supplies, tools, wrenches, and inspection equipment taken individually have relatively small development impact. Where possible they are developed concurrently with similar equipment required in the development program.

7.6 OTHER END ITEMS AND SPECIAL TASKS

Preliminary program planning estimates include one full-size mockup of the OTV engine to be fabricated and kept at P&WA/GPD. It will be updated by incorporating the latest engine design configurations and proposed configuration changes and engine handling equipment will be checked. A second mockup of the latest configuration is planned for delivery to the Government TBD months from go-ahead to be used for checkout of engine handling equipment, installation and procedures.

During the design of the OTV engine, the new parts required for preserving and packaging of the engine are designed. Service and field operations engineers provide their inputs to the design to assure that they are compatible with proposed GSE and handling equipment.

The reusability of these devices provides for a cost-effective program, since the engine assembly protection and certain packing devices will be used in the development phases to ship and preserve engines being transferred from the engine assembly area to the test facilities. The transfer of components for the development engines from the assembly area to the test facilities is also involved. Environmental protection will also be provided to the engines during transporting, handling, and storing of the components and engines to prevent mechanical damage and chemical deterioration that could jeopardize engine reliability.

All new preservation, packaging and shipping devices, for both components and the complete engine assembly, will be designed and tested during the development phases. Methods of handling and transportation should be established early to provide cost-effective protection for engines and support items over the program life cycle.

Services, hardware, and supplies required to conduct a potential capability and adaptability program for the OTV engine for alternate or future space missions are not estimated in these program planning estimates. Such estimates would be furnished should there be such requirements in an ongoing program. Task assignments would be in, but not limited to, the following areas:

1. Performance Investigations
2. Technology Investigations and Exploratory Tests
3. Vehicle and GSE Systems Investigations
4. Engineering Analysis and Feasibility Studies
5. Engine Solutions to Vehicle-Originated Problems.

7.7 VEHICLE DESIGN AND TEST COORDINATION

7.7.1 General

During the development of the Orbital Transfer Vehicle engine, questions and problems are expected to arise which will affect both the vehicle contractor in the development of the vehicle and the NASA in the operation of the OTV. The overall systems engineering approach needed to make the OTV an operational success requires the complete coordination of all matters that affect the total system. Engineering trade studies on the engine cannot be unilaterally settled but must take into consideration the interactions of the vehicle and the needs of the Government as the user. To ensure direct and accurate lines of communication, a team of engineering coordinators are planned for this effort in an ongoing program.

7.7.1.1 Change Coordination

Requirement Coordination Group field engineers will be in residence at the Government facilities and the vehicle contractor's facilities, and will contact these centers on a continuous basis to ensure a direct link with P&WA Government Products Division. Close coordination between resident engineers and the OTV Program is maintained through the Requirements Coordination Group and Program Engineering.

Assignment of a GPD in-house Interface Coordinator within the Requirements Coordination Group is planned to handle each center of field activity. He resolves problems associated with the Government operations center or vehicle contractors through the Product Support Engineer and appropriate Government vehicle personnel. The Interface Coordinator will be responsible for receiving problems or questions from the field and assigning internal action responsibility for rapid response to the field. He will also be responsible for action in the reverse direction when requirements exist for information from the field to support the OTV engine program.

In performing these tasks, the Interface Coordinator will work very closely with Program Engineering to ensure the proper response to both field and internal information requirements. Organizations such as Design, System Analysis, Materials, Integrated Logistics Support (ILS), Maintenance, and Reliability are the source of both the data to be supplied to the field and P&WA requirements for data. Program Engineering will work with the Interface Coordinators in both types of activity. The Interface Coordinator and Systems Engineering interact to enable data flow, assisting Program Management to make the required decisions with benefit of visibility of possible impact on vehicle and engine integration. When the need for change arises in the Interface Control Document or Engine Model Specification in the ongoing program, Program Engineering will initiate change proposals through a configuration control system.

7.7.1.2 Vehicle/Engine Integration

Of the many items requiring coordination between P&WA and the vehicle contractor, none is more important than the integration of the engine with the vehicle system. Though control documents specify many requirements and interfaces, past engine installation experience has required an effective coordination effort to maintain design and hardware schedule dates. The Field Engineers will coordinate the details of engine installation, including continual updating of the engine installation mockups at the vehicle contractor's plants. Minor configuration change coordination in either engine or vehicle that lead to a better and more maintainable engine installation is planned through the Field Engineer. During the ongoing program, engine checkout procedures and vehicle checkout procedures involving the engine, to be published in the contractor's technical publications, are to be supplied, reviewed, and coordinated as necessary by the Field Engineer.

A principal tool to be used to handle interface information will be the same Coordination Sheet procedure used during the initial study phases. It is used to transmit questions and replies between P&WA and the NASA or the vehicle contractor. The coordination sheet will also serve as a control document to make sure all requests are answered.

All P&WA support effort which would be supplied to the OTV program at operational sites will be under the direction of the P&WA Site Representative. The P&WA support team will be tailored to the needs of the program activity at the site and will supply both logistical and technical support.

Logistical support functions that are planned to be furnished in an ongoing operational program will include the warehousing and inventory control of support parts, performance of engine maintenance on installed engines, support for the operation of a site engine shop to perform second-level maintenance, operation and maintenance of engine GSE, and assistance to the vehicle contractor on engine installation and removal. P&WA would provide qualified personnel for engine inspection for quality control of engine work accomplished onsite or on engines and components received at the site. Field Service Representatives from P&WA, critical component vendors, and GSE specialists are planned to be part of the logistical team to provide technical advice, on-the-job training, and information for discrepancy reporting.

Technical support during operations will be supplied by Field Engineers, Performance Engineers, Instrumentation Engineers, and Data Reduction Analysts. Review of engine data from both functional and performance standpoints would be accomplished at the site or when more detailed analysis is required; the site technical personnel will transmit the data to the Development Engineering Group at P&WA/GPD. Results of P&WA/GPD investigations on either engine data or material from the operational site will be reported back to interested NASA and vehicle contractor personnel at the site by P&WA. Technical advice and assistance on engine installation, interface, and operational problems would be supplied by the P&WA Field Engineers.

Coordination of all engine activity at the site with the vehicle contractor and the NASA will be the responsibility of the P&WA Field Engineer. It is anticipated that prior to the start of operations at the site, an Associate Contractor Agreement (ACA) would be completed between P&WA and the vehicle contractor(s). This document would specify the areas of operation and task responsibilities in all areas of engine/vehicle interface, such as installed engine maintenance and reduction of engine flight data recorded on the vehicle on-board recorder. P&WA completed such agreements on previous programs and found it very helpful to use this approach in avoiding overlaps or oversights in interfacing areas of responsibility. By completing the ACA prior to the start of operations at a site, it would be possible for P&WA to specifically tailor the site support service and makeup to meet program needs.

7.7.2 Support Services

Some of the services that are planned to be supplied at each operational site by P&WA in an ongoing program are reviewed in the following paragraphs.

7.7.2.1 Field Level Modifications, Repair and Maintenance

Field level engine modifications, repair, and maintenance at the operational sites could be accomplished by P&WA technicians and inspectors under the supervision of the P&WA Field Service Representatives. Though engine technicians supplied by NASA, the vehicle contractor, or a third party are planned to be trained and supported by P&WA Service Representatives at the site, P&WA engine technicians and inspectors could be used at the start of operations and continue until operational capability has been established. This approach has an advantage during the ground and flight test program when engine maintenance procedures, operating procedures, and technical publications are being validated and revised. Technicians working directly for the engine manufacturer have direct lines of communication to report discrepancies and receive instructions for corrective action. Using this approach on the SR-71/J58 Program, P&WA was able to uncover and correct operational-type problems much sooner and more completely than is normal in a new vehicle program.

Maintenance on installed engines would be accomplished in keeping with the agreements completed with the vehicle contractor in the Associate Contractor Agreement. Maintenance on uninstalled engines and GSE would be accomplished in a site engine shop by the engine technicians and inspectors.

7.7.2.2 Checkout and Operation of Delivered Hardware

Assistance to the vehicle contractor in the checkout and operation of the engine will be provided by the engine maintenance team supported by a Field Service Representative and Field Engineer. Engine-oriented vehicle contractor personnel should attend one of the engine training programs prior to operations, but onsite training and technical assistance will be available from the P&WA personnel.

7.7.2.3 Removal and Replacement Documentation for Engine and GSE

Removal and replacement of engine and GSE components for maintenance or repair will be accomplished by the engine maintenance team using documentation as required and agreed to in the Associate Contractor Agreement and approved by the Government. Configuration accounting and documentation responsibility would be assumed by the engine maintenance team for items under their control for maintenance or repair.

7.7.2.4 Planning and Scheduling

Participation in engine and vehicle planning and scheduling activities for maintenance and operation would be the responsibility of the P&WA Site Representative.

7.7.2.5 Product Integrity Surveillance

Product integrity surveillance of associated engine, GSE, and vehicle interface tasks would be supplied by the inspectors in the engine maintenance team or in special cases by Field Engineers for all work accomplished by the engine maintenance team. On work accomplished by other than the engine maintenance team, quality control surveillance assistance would be supplied. Details of product integrity responsibility in engine vehicle interface areas would be spelled out in the Associate Contractor Agreement to avoid operational confusion.

7.7.2.6 Discrepancy Followup

Engine, or engine GSE material found discrepant, will be the subject of an investigation at the site by the Field Service Representative and Field Engineer. After completion of the local investigation, the material, with the investigation findings, will be forwarded to the Service Investigation Group at P&WA/GPD for further review, as necessary, with the OTV Development Engineering Group. Results of the findings of either an onsite or engine factory investigation of engine discrepant material would be reported to the Government. The same information will be supplied to the Government and vehicle contractor personnel onsite by the P&WA Site Representative.

7.7.2.7 Engine Flight Data Requirements

Engine-oriented communications, tracking, and data acquisition required by P&WA are minimal and listed in the preliminary Interface Control Document. Though there are about 14 selected parameters on the engine which will supply information on the engine's operation and condition, all of these can be stored on the Orbital Transfer Vehicle flight recorder for study after a flight. Special and extra engine instrumentation will be required for a few of the early flights during the operational program. The retrieval of data from this flight test engine instrumentation may be accomplished by an on-board magnetic tape recorder. Vehicle contractors generally expect to accept engine-supplied data on their on-board flight recorders and plan to be responsible for delivering the data reduced from recorder tapes, to the engine contractor after each flight. In keeping with the vehicle contractor plans, P&WA would not anticipate needing electronic data reduction equipment at the site location. If required, Data Analysts, Performance Engineers, and Field Engineers from the P&WA OTV Engineering Support Team would assist in processing the flight test data supplied by the vehicle data people after each flight and make the results available to the Government and vehicle engineers on the site.

SECTION 8

CONCLUSIONS

1. The RL10 and its derivatives are the only high performance upper stage engines that can be operational in the 1980's.

2. The RL10 derivative engine has a specific impulse within 4% of all OTV advanced engines.

3. RL10 derivative engines can provide the highest demonstrated reliability for OTV applications.

4. An expander cycle engine is inherently more reliable than a staged-combustion cycle engine. For example, the single burn demonstrated reliability of the advanced-expander cycle engine considered in this study is estimated to be 0.9967 (at FFC) while the staged-combustion cycle engine is estimated to have demonstrated a reliability of 0.9898 (at FFC).

5. There is no significant performance difference between advanced-expander and staged-combustion cycle engines.

6. The expander cycle engine provides as much potential for performance growth as a staged-combustion cycle engine for OTV applications.

7. A high-thrust expander cycle engine provides good performance and long life at low thrust operating conditions with no need for kitting.

8. There is a substantial difference in development cost between the advanced-expander cycle and staged-combustion cycle engines.

SECTION 9

RECOMMENDATIONS

1. Work should be initiated on RL10 derivative engines to support a 1987 OTV operational capability.*

2. Further OTV system definition work should be accomplished to better define engine/vehicle interface requirements and sensitivities in areas such as low thrust operating requirements and full thrust NPSH levels.

3. Component technology programs should be initiated in composite materials, the thrust chamber/nozzle and fuel turbopump areas leading to a data base on these critical items on which an advanced-expander cycle engine design can be based.

4. The staged-combustion cycle should be dropped from further consideration for advanced OTV applications due to its complexity, higher cost, lower reliability, and no significant performance difference when compared to an advanced-expander cycle engine.

*See P&WA FP80-807, RL10/RL10 Derivative Engine Program for the 1980's, 3 March 1980.